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ADVANCED SCOUT HELICOPTER (ASH) FLY-BY-WIRE FLIGHT CONTROL SYSTEM PRELIMINARY DESIGN

Volume I - System Design and Analysis

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Fort Eustis, Va. 23604

APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

This effort is one of two parallel contractual studies to evaluate the payoffs associated with application of advanced control technology (including fly-by-wire, fiber optics, and digital control laws to an ASH-sized helicopter. The associated study program under the title *Fly-by-Wire Versus Dual Mechanical Controls for the Advanced Scout Helicopter – Quantitative Comparison* (USAAVRADCOM Technical Report 80-D-10) was conducted by Bell Helicopter Textron under the terms of contract DAAK51-79-C-0007. As a parallel effort to these contracts, Sikorsky performed a similar study funded through their IR&D program. The results of the Sikorsky study may be made available to Government personnel by contacting the project engineer.

As a baseline for this study, the Medium Utility Transport (MUT) was chosen based upon similarity to ASH requirements and the use by MUT of a modern dual mechanical control system. The results of this study are useful not only for defining projected payoffs associated with the use of advanced control technology, but also for projecting a maturity rate for advanced control technology.

Mr. Joel L. Terry, Jr., Aeronautical Systems Division, served as project engineer for this program.

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This preliminary design study was conducted to assess the payoffs in applying fly-by-wire and other advanced flight control concepts to the anticipated Advanced Scout Helicopter (ASH). The study compares the advanced concept systems with a dual mechanical flight control system in the following areas: handling qualities, reliability, maintainability, availability, durability, survivability, EMP/EMI/lightning tolerance, cost, and weight.

The candidate vehicle for the study was the Medium Utility Transport (MUT) as defined in

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OVERVIEW

PURPOSE

This preliminary design study was conducted to assess the payoffs in applying fly-by-wire and other advanced flight control concepts to the anticipated Advanced Scout Helicopter (ASH). The study compares the advanced concept systems with a dual mechanical flight control system for the following parameters:

1. Handling Qualities
2. Reliability
3. Maintainability
4. Availability
5. Durability
6. Survivability
7. EMP/EMI/Lightning Tolerance
8. Cost
9. Weight

The candidate vehicle for the study was the Medium Utility Transport (MUT) as defined in USAAMRDL-TR-75-56A (Reference 1). This vehicle is a single rotor, composite fuselage design with a gross weight of 9,544 lb and useful payload of 960 lb. The results of this study were intended to support the configuration definition of the ASH program.

TECHNOLOGY BASIS

The technical portion of this study was carried out from 5 March 1979 to 28 March 1980 when the contractually required oral briefing was given in St. Louis. U.S. Army Program Management, Advanced Systems and Technology, and Applied Technology Laboratory personnel attended the briefing.

A ground rule of the contract was that chosen technology be suitable for program start in 1980.

For the study, the flight control system was defined to include: the cockpit controls, electronics, actuators, interconnecting links, and automatic control hardware. The electrical and hydraulic power supply characteristics were also included in the reliability and cost analyses.

1. Hoffstedt, D.J., and Swatton, S., ADVANCED HELICOPTER STRUCTURAL DESIGN INVESTIGATION, VOLUME I – INVESTIGATION OF ADVANCED STRUCTURAL COMPONENT DESIGN CONCEPTS, Boeing Vertol Co., USAAMRDL Technical Report 75-56A, Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, March 1976, AD A024662.

TASK BREAKDOWN

The work carried out in this study covered the following tasks:

Task 1 – System Requirements Definition

System requirements were based on the ASH Required Operational Capability (ROC) (Reference 2), applicable military specifications, Boeing Vertol experience on other programs (for example, the Tactical Aircraft Guidance System (TAGS), the Utility Tactical Transport Aircraft System (UTTAS), the Heavy Lift Helicopter (HLH)), and the results of simulation studies conducted as a part of the study. These requirements are summarized in the Flight Control System Candidates section and are defined in detail in Appendix A.

Task 2 – Candidate Configuration Definition

Research and simulation activities were conducted to establish the configuration in areas of potential risk, including the pilots' force controls, optical displacement transducers, and optical interconnect systems.

Candidate system configurations were synthesized and evaluated to provide a basis for more detailed study and hardware definition. The resulting optical and electrical system configurations are presented in the Flight Control System Candidates section. The research and simulation results are given in Appendix B.

Task 3 – Dual Mechanical FCS Design

The dual mechanical flight control system for the candidate vehicle described in Reference 1 was updated to provide the equivalent capability of the advanced candidates. Notable changes were the upgrading of the AFCS to provide a fail-operational capability, and further definition of system hardware based on UTTAS and LAMPS aircraft parameters. (See the Dual Mechanical Flight Control System section.)

Task 4 – Vendor Design Studies

System specifications as presented in Appendix A were prepared and preliminary design, cost and R/M information was solicited from qualified sources. The major activity centered on the controller, flight control electronics, and rotor control actuator design. Vendors supporting the study are shown in Table 1. Details on hardware designed for the study are given in the Detailed System Description section. Data considered proprietary by Boeing and its suppliers is presented in Volume II of this report.

Task 5 – Configuration Selection for Comparison with Dual Mechanical FCS

The original system candidates were updated as a result of the detail design study. Details on the configuration development are given in Appendix B. A fly-by-optics (baseline) candidate

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2. ADVANCED SCOUT HELICOPTER REQUIRED OPERATIONAL CAPABILITY (ASH) ROC) ATZQ-TSM-S, 11 January 1978 (CONFIDENTIAL).

TABLE 1. SUPPORT FOR FLY-BY-WIRE STUDY

SYSTEM ELEMENT	COMPANY
FIBER-OPTIC TRANSDUCERS	BOEING AEROSPACE COMPANY
SYSTEM ELECTRONICS	HONEYWELL AVIONICS DIVISION
MAIN/TAIL ROTOR ACTUATORS	BENDIX ELECTRODYNAMICS DIVISION
COCKPIT FORCE CONTROLLERS	LEAR SIEGLER ASTRONICS DIVISION
FIBER OPTIC CONNECTORS	ITT CANNON, AMPHENOL
FIBER OPTIC CABLE	GALILEO ELECTRO-OPTICS
ELECTRICAL CABLE	RAYCHEM
ELECTRICAL CONNECTORS	ITT CANNON, DEUTSCH, PYLE NATIONAL

and a fly-by-wire (alternate) candidate were selected. These are summarized in the Flight Control System Candidates section. Both candidates utilize a two-axis sidearm force controller with separate collective pitch controls and directional pedals. Processing for both primary and automatic control functions is digital. The fly-by-wire candidate uses dedicated wiring.

Task 6 – Comparative Analyses

The baseline and alternate candidates were compared with the dual mechanical system for the parameters shown in Table 2 and are detailed in the Comparative Analyses section.

RESULTS

Table 2 summarizes the results of the study. As shown, both the fly-by-wire and fly-by-optics candidates are superior to the dual mechanical in most respects. The fly-by-optics has further advantages over the fly-by-wire in EMP/EMI lightning tolerance, weight, maintainability, and availability. Production acquisition and operation/maintenance costs are similar for all candidates. Some program risk would be attached to the application of optical technology for a 1980 program start. Further development is needed in the area of fiber-optic transducers and interfacing hardware. The recent decision to delay the development of ASH will reduce this risk.

The application of digital technology to flight-critical control functions offers many advantages in simplification of the system hardware and its interfacing. These advantages are obtained in particular where passive direct digital output transducers such as those defined in this study can be used. However, these hardware advantages can be nullified if the software compromises flight safety. This can happen if the software complexity increases to the point where validation with the confidence needed to assure flight safety cannot be achieved.

The multiprocessor approach defined in this study provides hardware separation of the flight-critical functions (primary control) from other noncritical functions (automatic control). This approach limits the size and complexity of the flight-critical software and allows operation of the system without being compromised by backup systems.

Boeing recommends that emphasis in the continuing digital/optical studies be placed on optical transducer/interface development and multiprocessor system concepts so that a simple digital optical FCS which can be validated with the confidence needed for primary control will be developed.

TABLE 2. ANALYSIS SUMMARY

PARAMETER	DUAL MECHANICAL	COMPARED TO MECHANICAL FBW	FBO
HANDING QUALITIES		X	X
RELIABILITY			
FLIGHT SAFETY	MEETS REQUIREMENT	X	X
MISSION		X	X
MAINTAINABILITY		X	X (BEST)
AVAILABILITY		X	X (BEST)
DURABILITY		X	X
SURVIVABILITY - SMALL ARMS		X	X
EMI/EMP/LIGHTNING TOLERANCE	X (BEST)		X
COSTS PRODUCTION LIFE CYCLE	NONE HAS A SIGNIFICANT ADVANTAGE		
WEIGHT		X	X (BEST)

X ADVANTAGE

FLIGHT CONTROL SYSTEM CANDIDATES

SYSTEM REQUIREMENTS

The following is a summary of the requirements and ground rules established for this study. The complete system specifications are given in Appendix A.

General

1. The flight control system includes the following: cockpit controls, main rotor and tail rotor actuators, connecting hardware from controls to actuators, automatic flight control system hardware, and control stick and force feel system hardware.

2. All candidate systems shall have a primary flight control system (PFCS) that will allow flight with handling characteristics adequate to provide a get-home capability.

3. All candidates shall have an automatic flight control system (AFCS) that will provide stabilization in four axes and mission-related selectable functions. The interface between the PFCS/AFCS shall be designed to limit PFCS response to any AFCS failure including a simultaneous multichannel failure such that a 1-second delay and adequate trim margin is maintained after failure.

4. All candidates shall meet the Army flight safety reliability goals specified for this study; i.e., no more than one loss in 10 million hours based on a mission time of 1 hour.

   Mission and maintenance reliability predictions of this study shall be used to rank the designs relative to the Army-specified goals of one abort per 10,000 hours and one maintenance action per 2,500 hours.

5. All candidates shall be supplied from a dualized hydraulic supply, backed up with a third supply of limited capability.

6. All candidates shall meet the requirements of the Advanced Scout Helicopter (ASH) Required Operation Capability (ROC) (Reference 2).

7. The system shall also conform to the requirements of MIL-F-9490D (Reference 3).

8. Electronic equipment shall conform to the requirements of MIL-E-5400T (Reference 4).

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3. Military Specification, MIL-F-9490D, FLIGHT CONTROL SYSTEMS DESIGN, INSTALLATION AND TEST OF, PILOTED AIRCRAFT, GENERAL SPECIFICATION FOR, Department of Defense, Washington, D.C., 6 June 1975.

4. Military Specification, MIL- -5400T, ELECTRICAL EQUIPMENT, AIRBORNE, GENERAL SPECIFICATION FOR, Department of Defense, Washington, D.C., 16 November 1979.

9. Hydromechanical equipment shall conform to the requirements of MIL-H-5440G (Reference 5).

Primary Flight Control System (PFCS)

- Mechanical System

  1. The system shall be similar to that described in Reference 1 for the Medium Utility Transport; i.e., dual mechanical controls from pilot/copilot to rotor boost actuators.

  2. The system shall incorporate the control driver actuators, a cyclic decoupler actuator, and AFCS actuators (integrated in the rotor actuator) necessary to interface with the AFCS.

- Nonmechanical Systems

  1. The cockpit controls shall be low-displacement, force type.

  2. To facilitate flight with AFCS OFF, the cockpit controls shall have a conventional arrangement (i.e., longitudinal/lateral control in right hand, collective pitch control in left hand, and directional pedals). Shaping of cockpit control inputs in the PFCS shall be used to enhance control under the AFCS OFF condition.

  3. System channels shall be in-line (self) monitored.

  4. System electronics shall interface with a hydraulically dualized actuator.

  5. Use of stabilization sensors as part of the PFCS shall be minimized.

Automatic Flight Control System (AFCS)

1. Stability and control characteristics shall be tailored to minimize pilot fatigue for the night, nap-of-the-earth mission.

2. To meet the desired operational characteristics, the AFCS shall be fail-operational/fail-safe for basic stability and control modes, and fail-safe for mission-related selectable modes (i.e., hover hold and altitude hold).

3. System architecture shall allow performance of the night, nap-of-the-earth mission with minimum upset on first failure.

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5. Military Specification, MIL-H-5440G, HYDRAULIC SYSTEMS, AIRCRAFT, TYPES I AND II, DESIGN AND INSTALLATION REQUIREMENTS FOR, Department of Defense, Washington, D.C., 14 September 1979.

4. For normal operation, the AFCS shall provide Level 1 flying qualities per MIL-F-83300 (Reference 6). Level 2 shall be allowed with certain sensor failures. A get-home capability shall be provided with PFCS alone. Detailed configuration of the AFCS shall be developed by pilot-in-loop simulation. Basic characteristics shall be as shown in Table 3.

5. The system shall be designed to achieve maximum integration with avionics sensors where this integration does not compromise the host system or the AFCS.

NONMECHANICAL SYSTEM CANDIDATES

This section describes the candidate systems configured in accordance with the requirements defined in the previous section. Details on the development of these candidates, starting with the configurations originally proposed, are given in Appendix B.

A digital/optical approach was selected as the baseline candidate. As noted previously there is some risk associated with the optical technology relative to a 1980 program start. Because of this an alternative fly-by-wire approach is also presented.

Fly-by-Optics

The following is a brief description of the baseline fly-by-optics system. Details of the system design are given in the Detailed System Description section.

The baseline system is illustrated in Figure 1, which shows the candidate vehicle with the digital/optical system installed. The mechanical run to the tail rotor actuator is retained to minimize vulnerability of the hydraulic lines in the tail boom. Figure 2 is a block diagram of the baseline digital/optical flight control system. The system incorporates optical transducers for pilots' inputs and rotor actuator position feedback. Main rotor actuators are controlled by electrical signalling. The use of electrical control precludes the need to provide signal conversion electronics on the actuators. External stabilization sensors used by the AFCS, including the vertical gyro, turn rate gyro, and nonredundant sensors (Figure 3), are shared with the pilots' and other avionics systems. The flight control system provides an airspeed sensor in each flight control processor and a barometric altitude sensor in the sensor multiplex/test interface unit.

- Principal Components

Low displacement (force-type) pilot's controls (with optical displacement transducers) overcome limitations imposed by the mechanically synchronized controls used in the systems designed for HLH and UTTAS, by avoiding single-point jam failures. The flight safety reliability of these systems was previously dominated by these failure modes.

The Flight Control Processor (FCP) is fully digital. It includes separated dual microprocessors for the flight safety critical primary control functions and a third microprocessor for the

6. Military Specification, MIL-F-83300, FLYING QUALITIES OF PILOT V/STOL AIRCRAFT, Department of Defense, Washington, D.C., 31 December 1970.

TABLE 3. AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) CHARACTERISTICS

AFCS	AXES				REDUNDANCY
	LONGITUDINAL	LATERAL	DIRECTIONAL	VERTICAL	
BASIC MODES					
TRIM HOLD	X	X	X		FAIL OP/FAIL SAFE
CONTROL RESPONSE					
ATTITUDE	X	X			
RATE			X	X	
SELECTABLE MODES					FAIL SAFE
ALTITUDE HOLD				X	
GROUND VELOCITY HOLD AND RESPONSE	X	X			

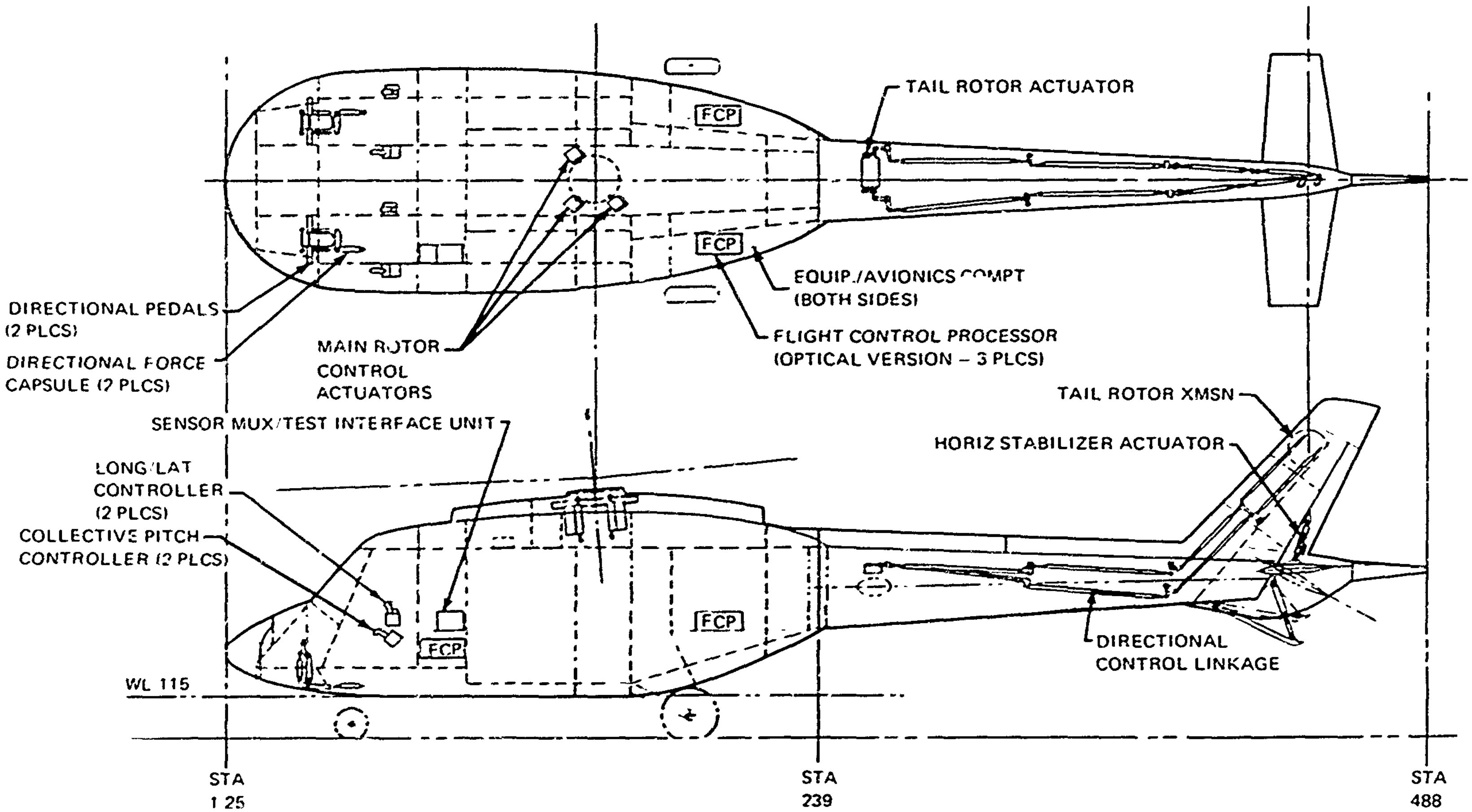


Figure 1. Fly-By-Optics Control Arrangement - ASH Study.

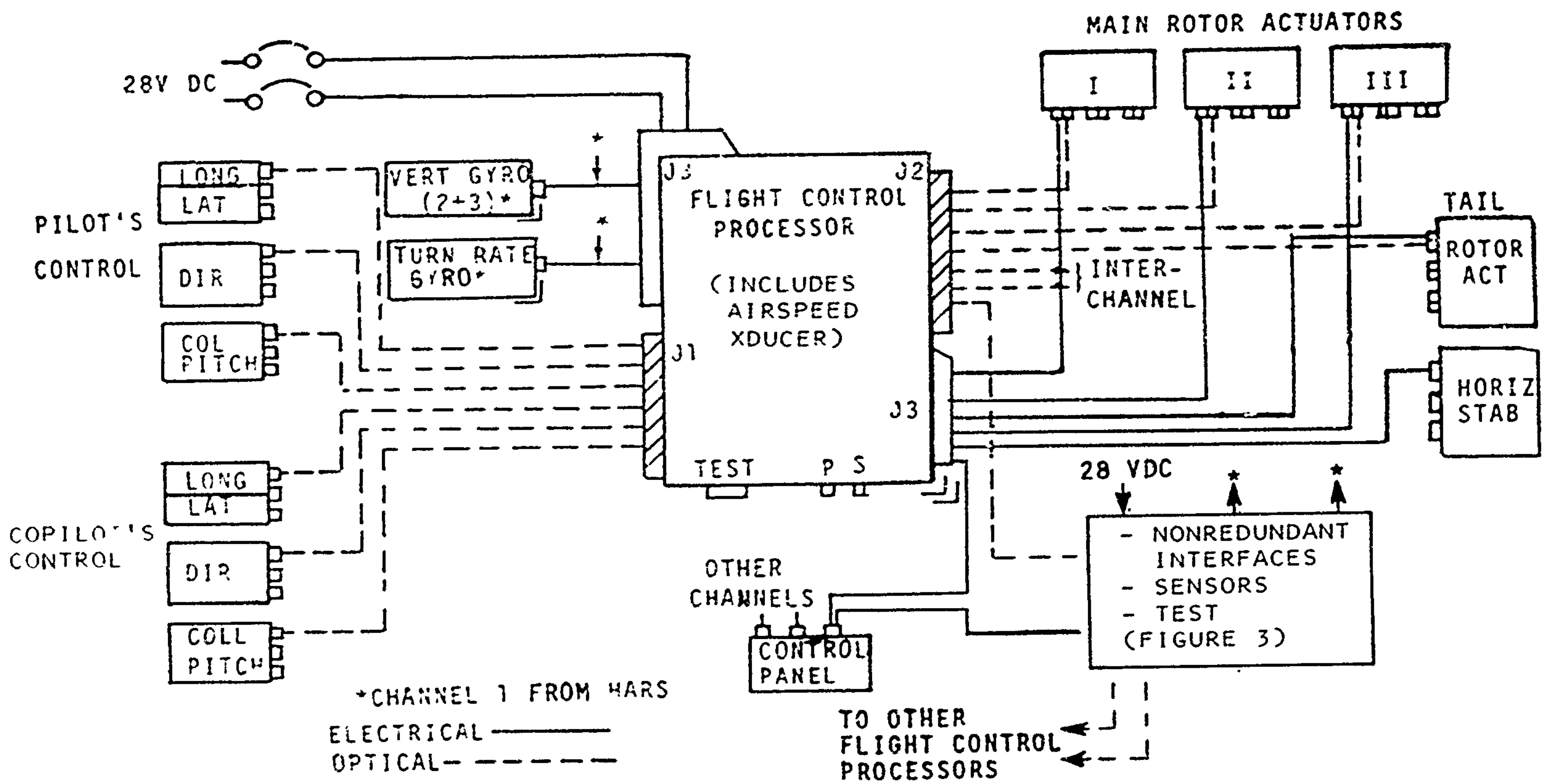


Figure 2. Baseline ASH Flight Control System - Equipment Diagram.

automatic flight control functions. The FCP is housed in a single line-replaceable unit (LRU). Figure 3 defines the interface with nonredundant sensors and the test interface function used to check the system and isolate failures. The nonredundant sensor set used in the YAH-64 Advanced Attack Helicopter has been assumed for this study.

The Rotor Control Actuators are of an integrated design including a control stage interfaced with the dualized primary flight control processors and a power stage, which includes a conventional dualized control valve (driven by the control stage) and an output ram. The two control loops are closed by optical position transducers.

- Redundancy Management

In-Line (Self) Monitoring The in-line monitored concept of redundancy management is used for the primary system and involves the use of three channels with two identical signal paths in each channel between the cockpit controls and the actuator input (Figure 4). If a discrepancy occurs which is greater than a preestablished tolerance level, that channel is considered to have failed and is shut down. The electrohydraulic actuators have dual hydraulic sections and triplex electrical sections.

Each channel is powered by an independent electrical supply. Tracking of three channels is maintained by control of overall gain tolerances. Gain control is enhanced by use of digital transducers and processing. Channel inputs are summed magnetically in the electrohydraulic valves of the actuator.

Inherent fail safety without time-critical switching is maintained for first failures by use of the magnetic summing. The baseline primary system was configured with the following levels of redundancy.

1. Sensors (optical path), PFCS signal processing and PFCS electro-optical interconnect lines: Dual-fail operative.

2. Mechanical portions of sensors: Single-fail operative checked in a background interchannel comparison.

3. Hydromechanical portions of the actuators and hydraulic lines to actuators: Single-fail operative.

4. Electrical power supply: Dual-fail operative.

5. Hydraulic power supply: Dual with switched limited capacity third channel for backup.

Fly-By-Wire

The alternate fly-by-wire candidate (Figure 5) closely parallels the baseline (fly-by-optics) candidate in overall structure. The principal differences lie in the use of analog linear variable

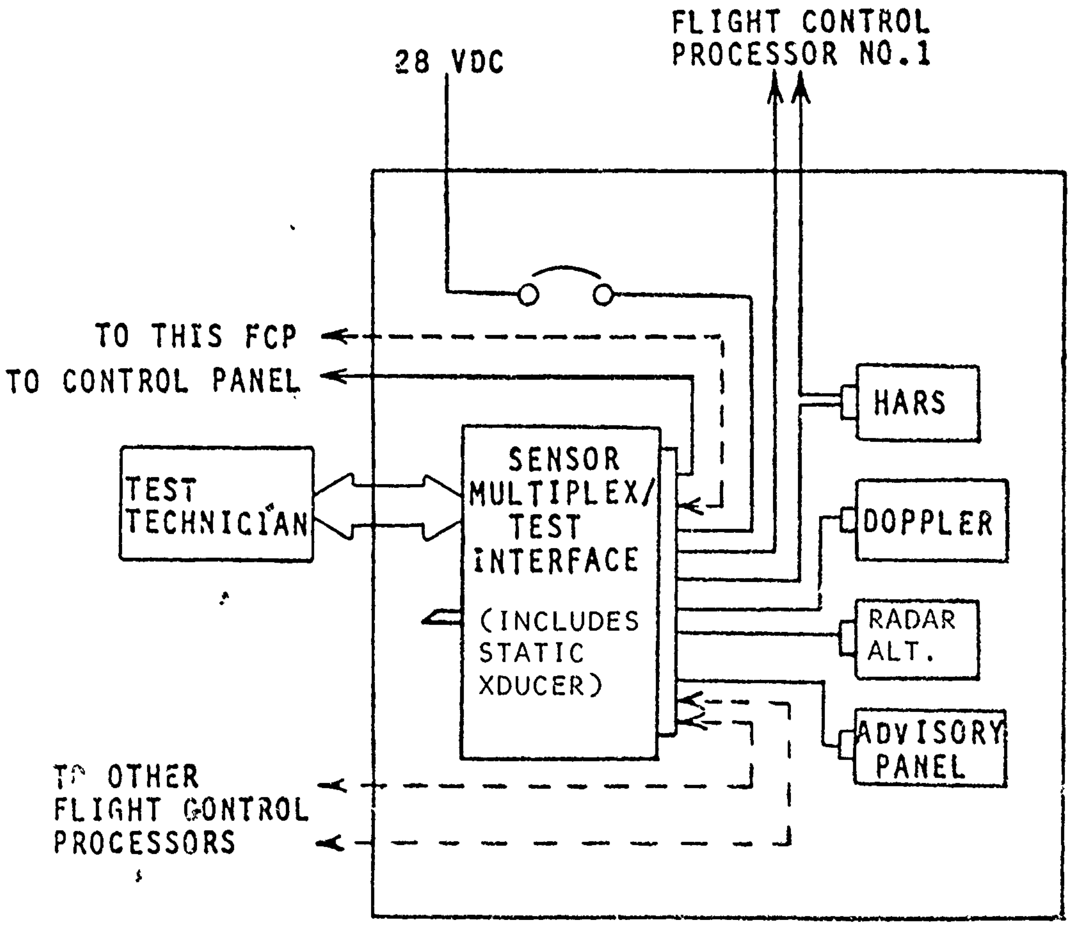


Figure 3. Nonredundant Interfaces - Sensors/Test.

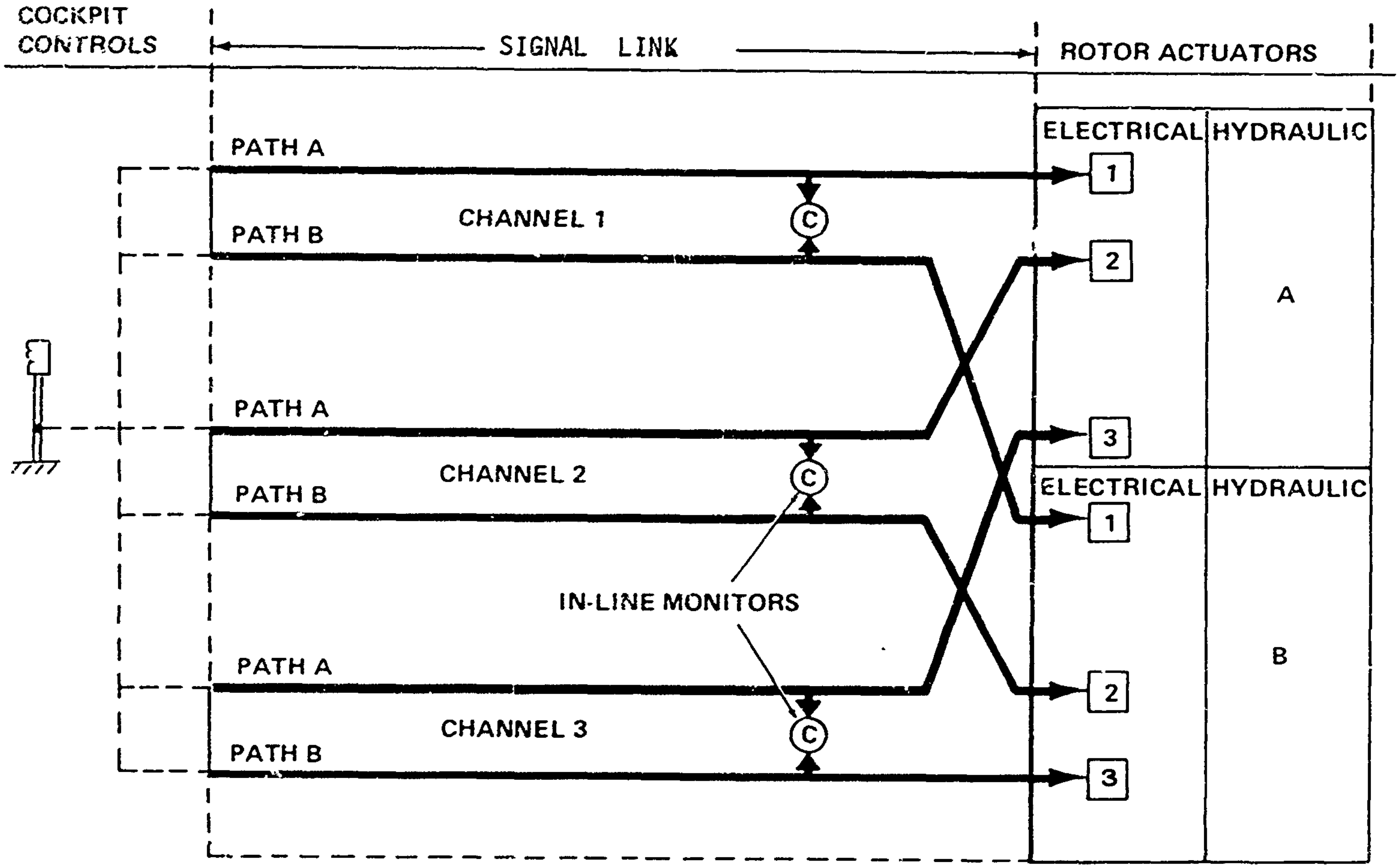


Figure 4. Redundancy Management Concept.

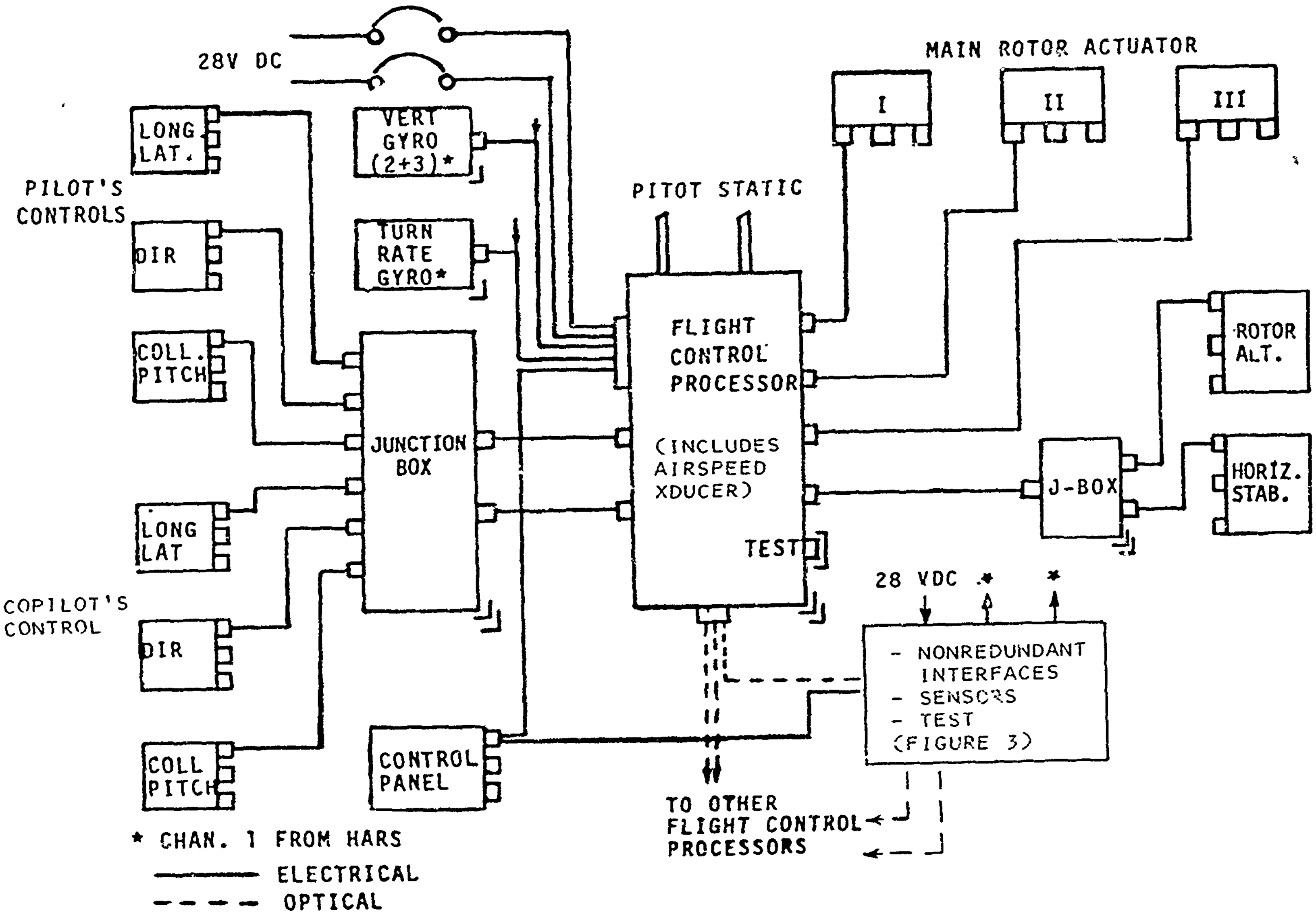


Figure 5. Alternate ASH Flight Control System - Equipment Diagram.

differential transformers (LVDT) for cockpit control position sensing and actuator position feedback, and dedicated multiconductor cabling in place of optical cable to connect these transducers to the flight control processor (FCP).

The concept of the point-to-point connection without branching as used in previous designs (HLH and UTTAS) means that junction boxes are needed to collect signals in an area for return to the FCP. This approach minimizes the number of connectors required on the FCP.

The system would incorporate double cable shielding to meet the EMI/EMP levels defined for the Control Media study (Reference 7). Use of this type of cable is assumed in cost and weight analyses.

The pilots' controllers and rotor actuators are functionally similar to those of the baseline except for the position transducers used. For fly-by-wire, the FCP incorporates analog-to-digital convertors in place of the optical-to-digital convertors provided in the baseline candidate.

DUAL MECHANICAL FLIGHT CONTROL SYSTEM

The dual mechanical flight control system of the Medium Utility Transport (MUT) helicopter, as described in Reference 2, was updated to provide the equipment detail necessary for comparison with the baseline and alternate system configurations. Detail definition of system weight and cost was based on the Boeing UTTAS (YUH-61A) and LAMPS aircraft designs.

Overall Configuration

The equipment complement was upgraded to provide a system which is functionally equivalent to its nonmechanical competitors. This meant addition of a triplex fail/operative AFCS with hover and altitude hold modes in place of the dual SCAS provided in the MUT configuration. This system includes the additional electronics and actuators to provide the equivalent function. Figure 6 shows the system configuration as installed in the MUT fuselage. Figure 7 defines the system hardware and electrical interconnect.

Principal Components

The pilots' controls comprise a conventional displacement-type cyclic control stick, collective pitch lever, and pedals for directional control. The pilot's and copilot's individual controls are connected via jam override detents, which allow continued control with one side of the dualized linkage jammed.

Pilots' force feel is via conventional feel springs. Trim is maintained by control driver actuators in each axis. These actuators also respond to AFCS inputs.

7. MECHANIZATION STUDY REQUEST FOR PROPOSAL, DAAK51-79-Q-0129, Control Media, Applied Technology Laboratory, U.S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia, 1 November 1979.

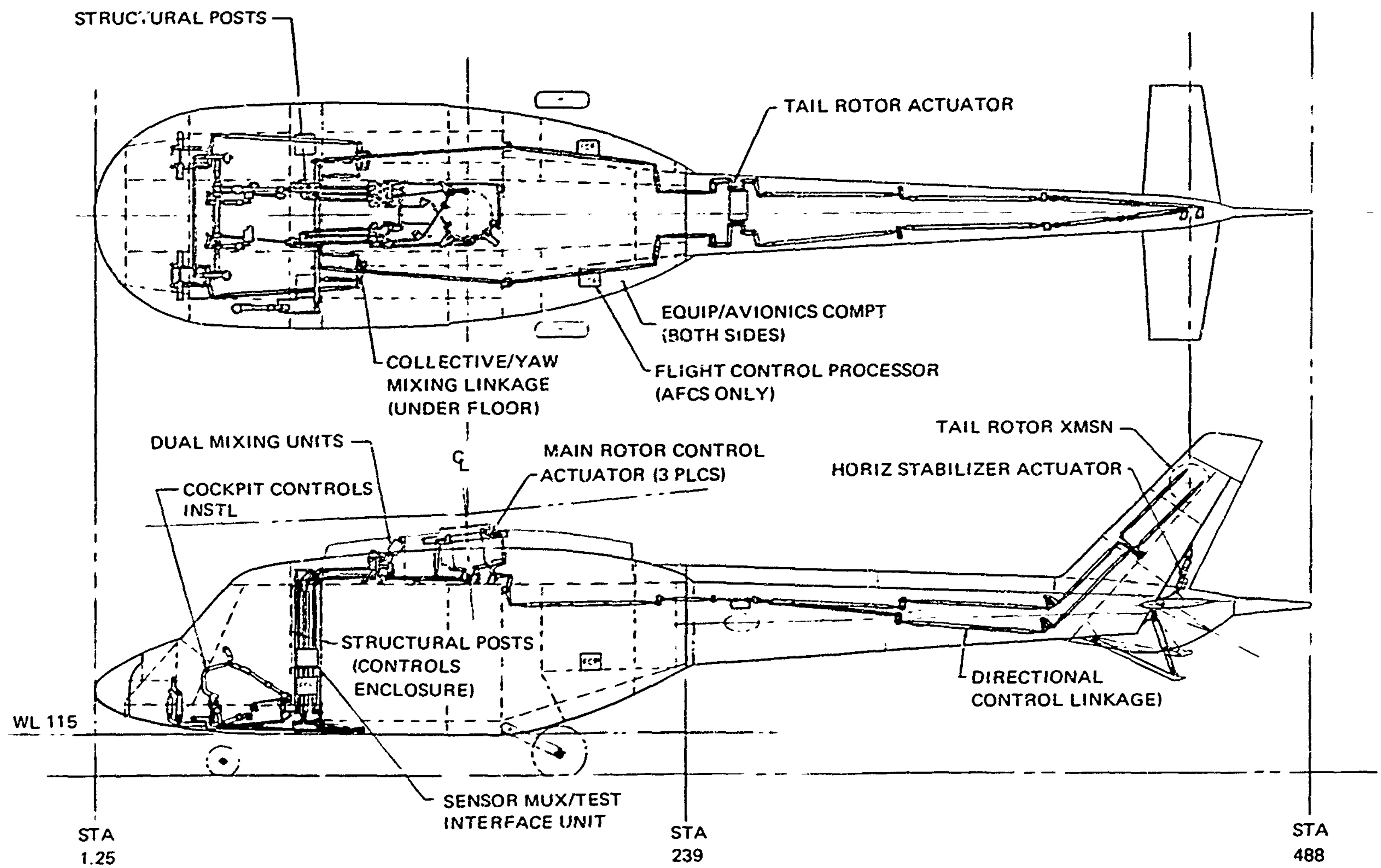


Figure 6. Dual Mechanical Controls Arrangement - ASH STUDY.

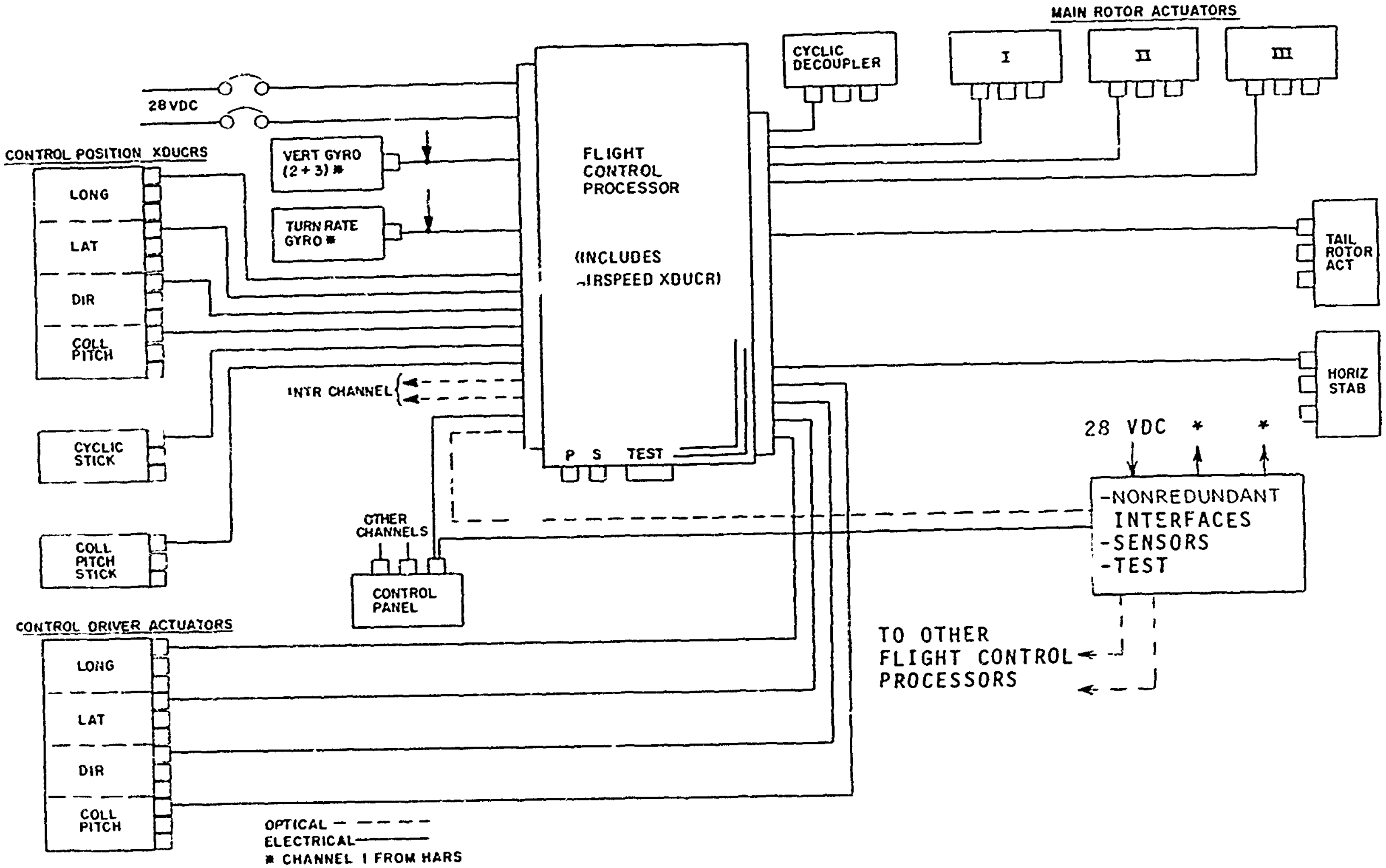


Figure 7. Dual Mechanical Flight Control System - AFCS Equipment Diagram.

Dual push rod systems are carried from the pilots' cyclic and collective pitch controls to the rotor control actuators via mechanical mixing units (which convert the axis motions to actuator motions needed for control).

The cyclic decoupler actuator located in the mechanical mixer provides slow rate cyclic inputs as a function of airspeed to compensate for control gradient nonlinearities over the airspeed range.

The rotor control actuators are of an integrated design. They accept inputs from both the pilot's and copilot's mechanical control runs. They include AFCS actuators which provide inputs that are functionally in series with the pilots' inputs. See the Dual Mechanical Flight Control Boost Actuator section for details.

The Flight Control Processor (FCP) is similar to that provided for the nonmechanical competitors, except that the circuitry devoted to the PFCS is omitted. The interface with stabilization sensors is identical to the other versions.

System Transfer Functions

Transfer function of the mechanical system are essentially identical to those shown in Appendix A (Figures A-7, A-8, and A-11), except that conventional displacement-type controls, a force feel system, and mechanical linkages with electromechanical actuators replace the limited displacement, force-type controls and equivalent electronic functions.

Redundancy Management

Failures of the dual mechanical linkage are handled in the following manner:

1. Open Failures – By virtue of the duality of the mechanical linkage there will be no degradation upon open failures of one side (pilot's or copilot's). Loss of control will normally be detected by inspection since the cockpit controls are crosstied allowing either pilot to use the remaining linkage path.

2. Jam Failures – Jams in the pilots' controls and linkage are overcome by breakout of the cockpit control crosstie, which then allows control via series spring capsules at the actuator input. In this case control authority and gradient is reduced. A standing force must be applied to overcome the actuator spring force.

DETAILED SYSTEM DESCRIPTION

The following sections give additional details on the candidate systems in terms of:

1. Cockpit Controls
2. Flight Control Electronic Units
3. Rotor Control Actuators
4. Redundancy Management
5. Power Supplies

COCKPIT CONTROLS

Functional Arrangement

Figure 8 shows the recommended arrangement of the cockpit controls for the FBW/FBO systems. As noted in Appendix B, Boeing has evaluated this arrangement on the moving base simulator.

The approach uses a two-axis hand controller for longitudinal/lateral control, single-axis hand control for collective pitch, and conventional rudder pedals that are connected to a low displacement force transducer for directional control. A trim display, located on the instrument panel between the two pilots, defines the axis trimmed position relative to its extreme limits.

Force/displacement characteristics of the controls are given in Table 4. These have been established based on simulation results, the results of the F-16 controls optimization, and flying qualities specifications (i.e., MIL-H-8501A (Reference 8) and MIL-F-83300). Vernier trim for longitudinal/lateral axes and the trim command button for longitudinal, lateral, and directional forces are provided on the right-hand controller. Directional/collective vernier trim and the collective force trim trigger are provided in the left-hand controller.

Hardware Description

- Longitudinal/Lateral Control

Figure 9 shows the Lear Siegler outline drawing for the two-axis longitudinal/lateral controller. Details of this proprietary design are given in Volume II of this report. This design will allow variation of force characteristics with minimum cost impact. The original design for the F-16 prototype incorporated a virtually zero displacement force transducer (i.e., ± 0.030 in. at grip reference position). Subsequent refinement for the production F-16 showed that a ± 0.200-in. displacement was desired for longitudinal control and a ± 0.100-in. displacement was desired for lateral control. The original design was modified by inclusion of a displacement multiplier in the grip (while retaining the original low displacement transducer).

8. Military Specification, MIL-H-8501A, HELICOPTER FLYING AND GROUND HANDLING QUALITIES, GENERAL REQUIREMENTS FOR, Department of Defense, Washington, D.C., 3 April 1962.

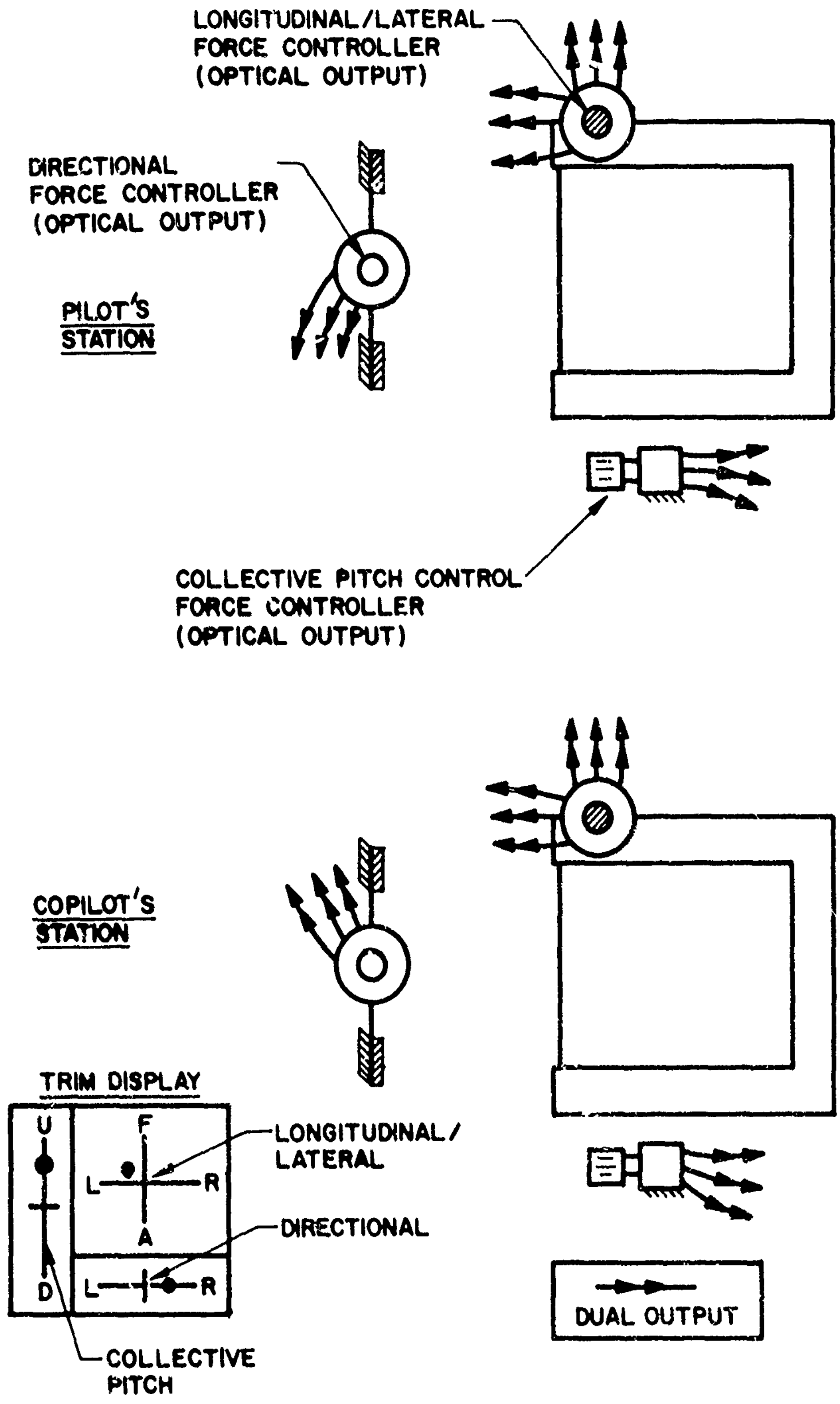


Figure 8. Baseline Cockpit Control Concept.

TABLE 4. COCKPIT CONTROL FORCE/DISPLACEMENT RANGE

FUNCTION	FULL-SCALE RANGE
LONGITUDINAL	$\pm$ 20 lb $\pm$ 0.200 in.
LATERAL	$\pm$ 10 lb $\pm$0.100 in.
DIRECTIONAL	$\pm$ 30 lb $\pm$ 0.50 in.
COLLECTIVE	$\pm$ 10 lb $\pm$ 200 in.

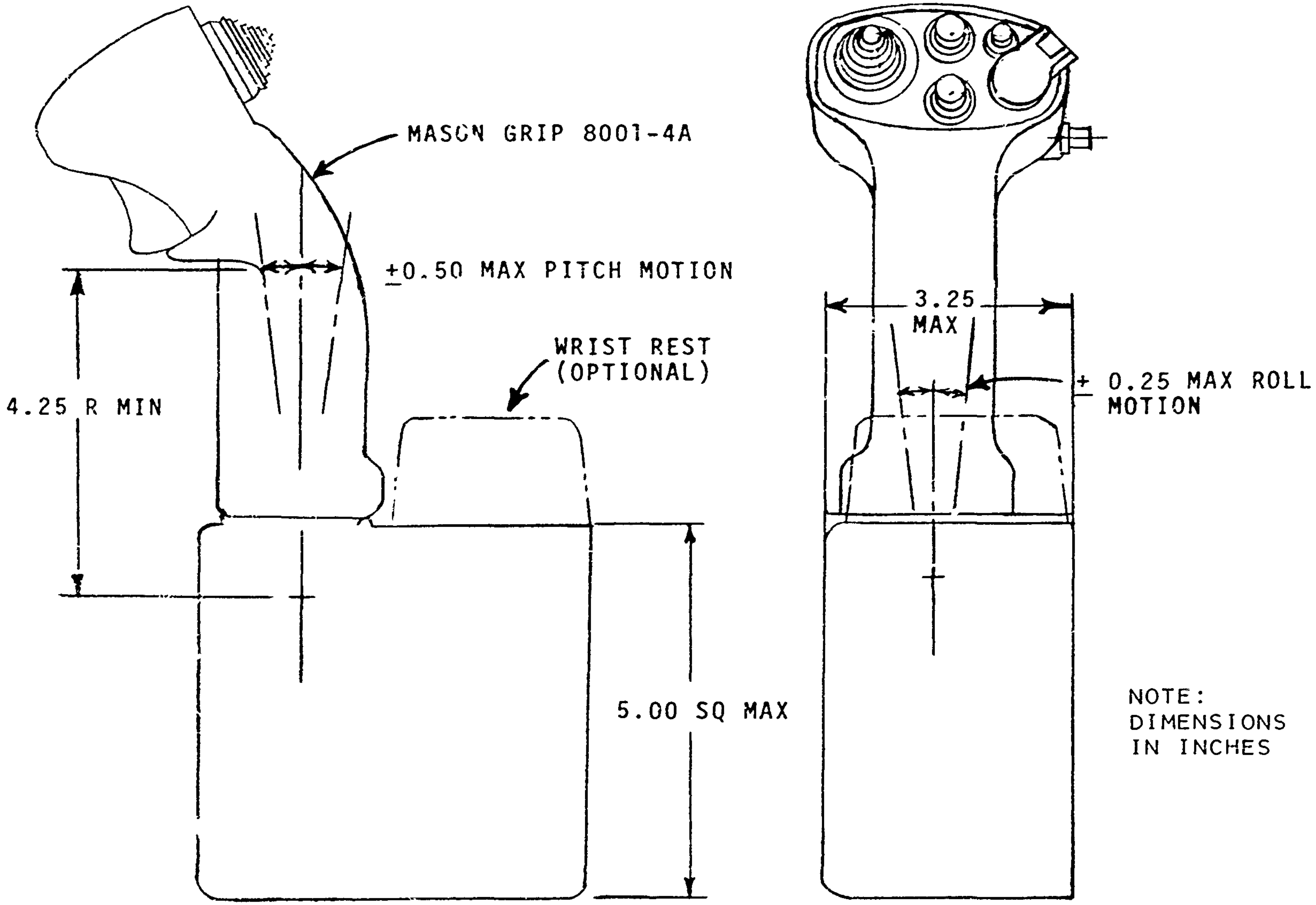


Figure 9. Longitudinal/Lateral Controller - Outline.

In configuring a controller that uses a digital transducer, it is desirable to increase the position transducer displacement in order to maximize the linear size of the least significant bit. The desired displacement and force characteristics could not be achieved within the constraints of the F-16 design. The configuration shown in Figure 9 was developed. In production, this unit will be lower in cost than the F-16 controller.

The longitudinal/lateral controller would be integrated with a right-side arm rest that could allow for forward/aft and vertical adjustment. This is a complication and was not required on the F-16. The arm rest would drop to a stored position for entrance/egress on the right side.

- Collective Pitch Control

The collective pitch controller will be similar to the longitudinal/lateral, but without lateral freedom. It would be located low at the left side of the seat in the conventional mid-to-low collective position and would be oriented for a predominantly up/down input. In this position it may not be necessary to provide stowage for entrance/egress on the left side.

- Directional Control

The pedals will be conventional and similar to the arrangement provided for the mechanical candidate. Each pedal set will be restrained by a force transducer (Figure 10). The precision spring capsule incorporates two springs preloaded to the output member and having zero breakout force (an electronic breakout is provided in the flight control processor). To minimize hysteresis effects, the spring ends are supported in a self-aligning joint to make up for spring end squareness variation.

If necessary and desired, two additional force capsules as described above could be applied to serve a wheel brake controller. These would be placed on the pedals in place of the brake cylinders and interfaced into a brake-by-wire system via the flight control processor The hardware to accomplish this has not been included in the flight control system cost and weight data.

- Control Grips

The control grips contain optical switches and circuitry necessary to interface with the optical excitation and output lines. These switches would be a new development. Boeing has discussed development of the grips, including optical switches, with Mason Electric.

Figure 9 shows the Mason design for the cyclic grip. The grips could be made from aluminum castings. To minimize losses in connection, the grips would be supplied with an optical pigtail which would be interfaced with an optical terminal board within the controller.

FLIGHT CONTROL ELECTRONIC UNITS

Flight Control Processor Functional Description

As noted previously, Boeing Vertol recommends use of a multiprocessor architecture for the flight control processor. In contrast to the single processor approach (for example, the F-18 Flight Control System), this configuration limits the scope of software that can affect flight

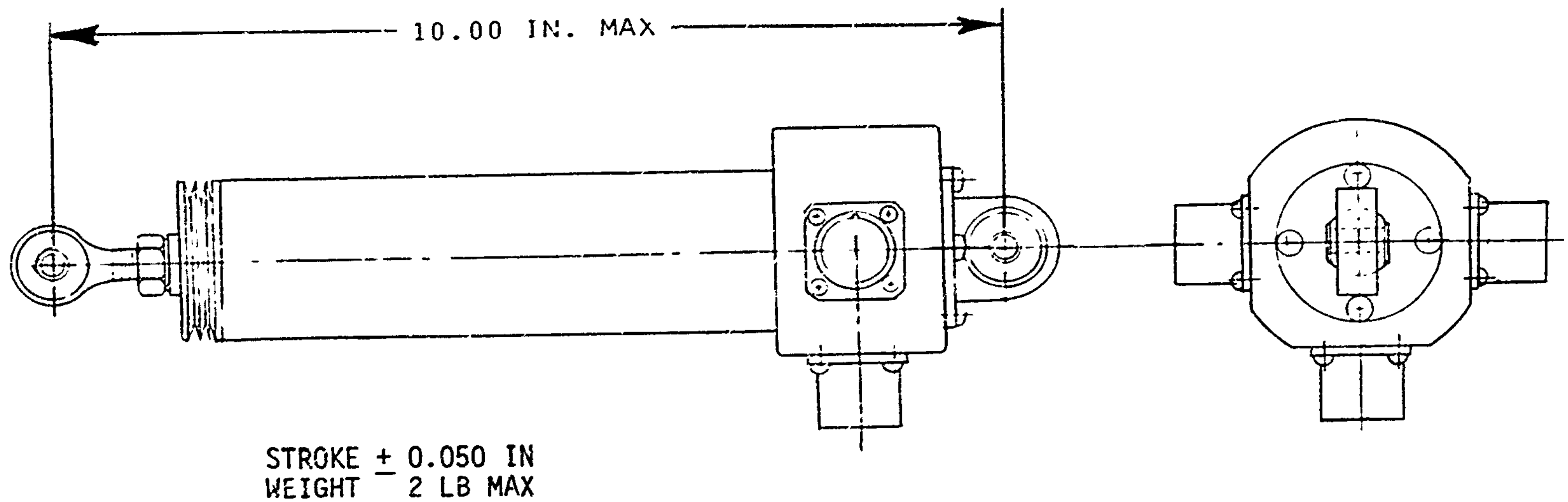


Figure 10. Pedal Force Transducer.

safety to the primary control functions (approximately 3K of memory). In this way, Boeing expects to achieve software validation with a high degree of confidence.

This multiprocessor approach also enhances failure coverage by direct comparison of two independent computations rather than self-check of a single processor. Boeing's triplex multiprocessor approach achieves the failure coverage of a quad cross-channel monitored system without the danger of cross-channel failure propagation and without the complication of a quad voter. Less than 20 percent of program memory is used for redundancy management and BITE.

Figure 11 is a block diagram for the flight control processor showing the multiprocessor approach recommended by Boeing Vertol. A common clock serves Path A, Path B, and the AFCS. The PFCS (Path A and B) processors within a channel run synchronously whereas the AFCS processors (working cross-channel) are frame synchronized.

Fiber optic excitation is provided to the pilot's controls and actuators in a rotating sequence; sensor data is updated 160 times per second. Each sensor provides a dualized return which is processed in the sensor convertor and placed in the memory of its path processor. (Details on the proprietary sensor and interface are given in Volume II.)

The PFCS processor computes rotor actuator current commands using the sensor inputs and triplex AFCS commands which are generated locally as well as by the adjacent two channels. The functions of the PFCS processor are:

1. Excitation of sensors; receipt and filtering of sensor signals.
2. Adjustment and storage of trimmed position.
3. Receipt and distribution of interchannel signals.
4. Voting and failure monitoring of AFCS axis commands.
5. Rate and authority limiting of selected AFCS commands.
6. Interface of limited AFCS signals with PFCS position commands.
7. Limiting and mixing axis command signals.
8. Converting actuator commands to analog format.
9. Failure detection and disconnect.

Figure 12 is an information flow diagram for the processor as mechanized by Honeywell, Incorporated. The processor employs Intel 8086 16-bit microprocessors to perform the PFCS and AFCS computation functions.

Electronic Unit Hardware Details

Table 5 summarizes the physical characteristics of the electronic units described in the following paragraphs.

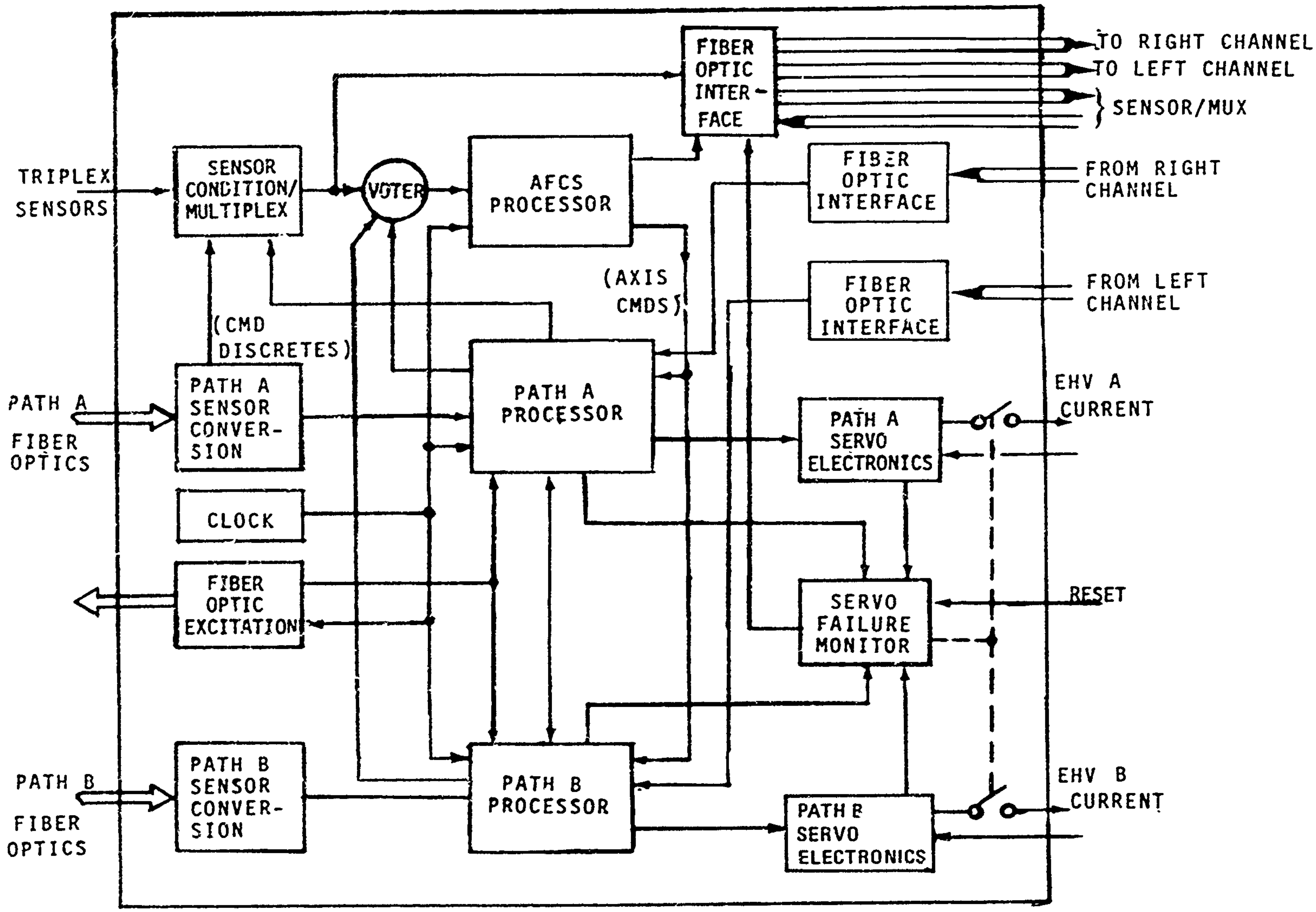


Figure 11. Baseline Flight Control Processor - Block Diagram.

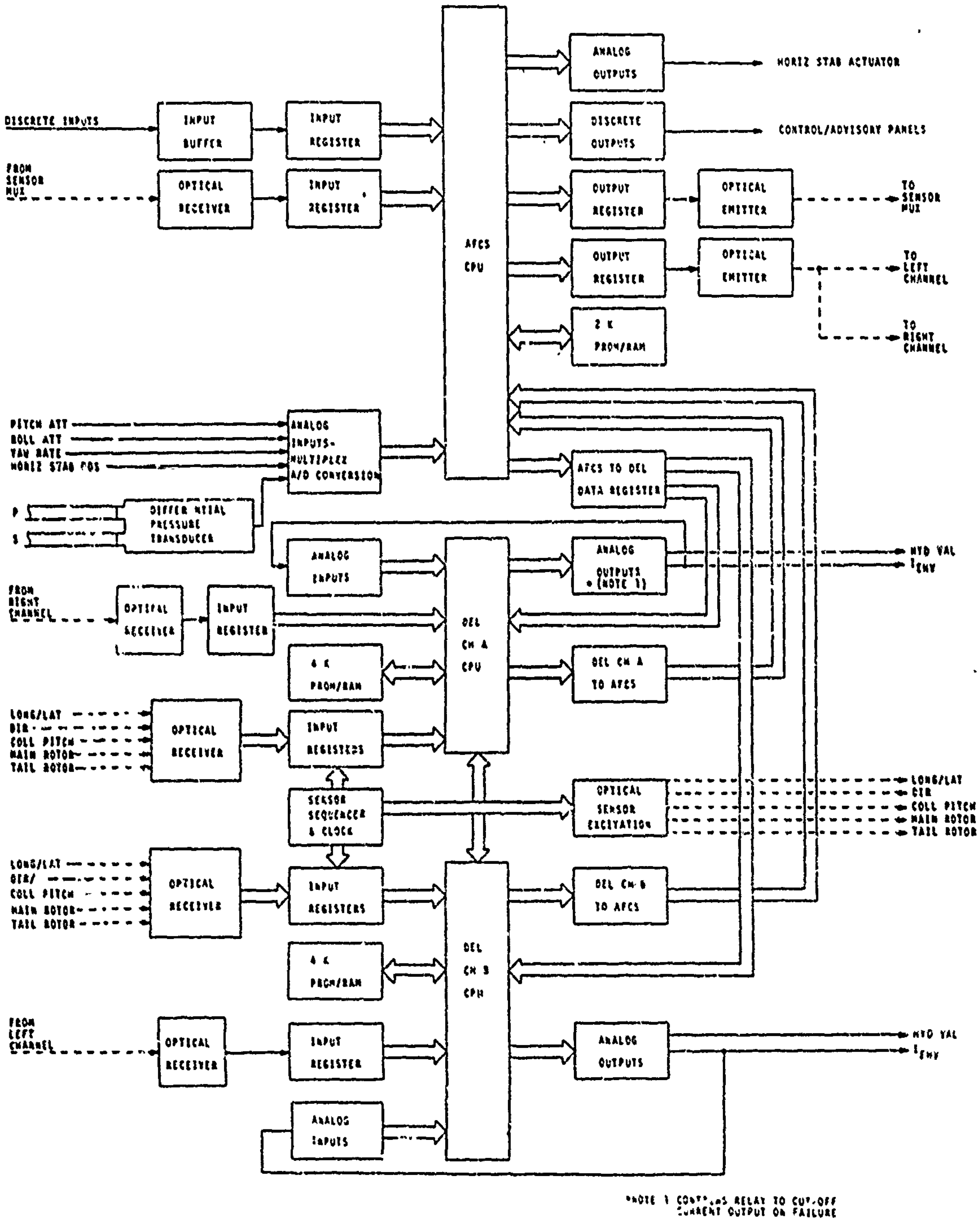


Figure 12. Baseline Flight Control Processor - Information Flow Diagram.

TABLE 5. ELECTRONIC COMPONENT COMPARISONS

PARAMETER	FLIGHT CONTROL PROCESSOR			SENSOR MUX/ TEST INTERFACE UNIT	CONTROL PANEL
	BASELINE	ALTERNATE	DUAL MECHANICAL		
Size (in.)	7.5 x 18.0 x 7.62	7.5 x 16 x 7.62	7.5 x 12.3 x 7.62	7.6 x 117.62	3.75 x 5.75 x 4.45
Weight (lb)	25.7	22.9	19.0	15.8	2.5
Electrical Power	159W	150W	98W	82W	17W
Failures/ Million Hours	440	290	290	188	75

- <u>Flight Control Processor</u>

Figure 13 shows the baseline flight control processor (fly-by-optics). A 3/4 ATR chassis provides the enclosure for:

1. Twelve plug-in card assemblies
2. Airspeed transducer
3. Inverter/filter assembly
4. EMI module
5. Fiber-optic transmitter/receiver module

The plug-in card assemblies are 6.25 in. x 6.24 in. fiberglass boards containing the state-of-the-art digital and linear microcircuitry.

The EMI and fiber-optic modules are separate assemblies mounted in the rear near their respective connectors. The EMI module is located in the lower one-third of the rear compartment near the electrical connector. The fiber-optic module is located in the upper two-thirds of the rear compartment; the optical connectors are an integral part of the module. All the fiber-optic transmitting and receiving circuitry is contained within the module. The chassis is constructed to allow the use of cooling air.

Figure 14 shows the <u>alternate</u> flight control processor (fly-by-wire). This unit is identical to <u>baseline</u> except for:

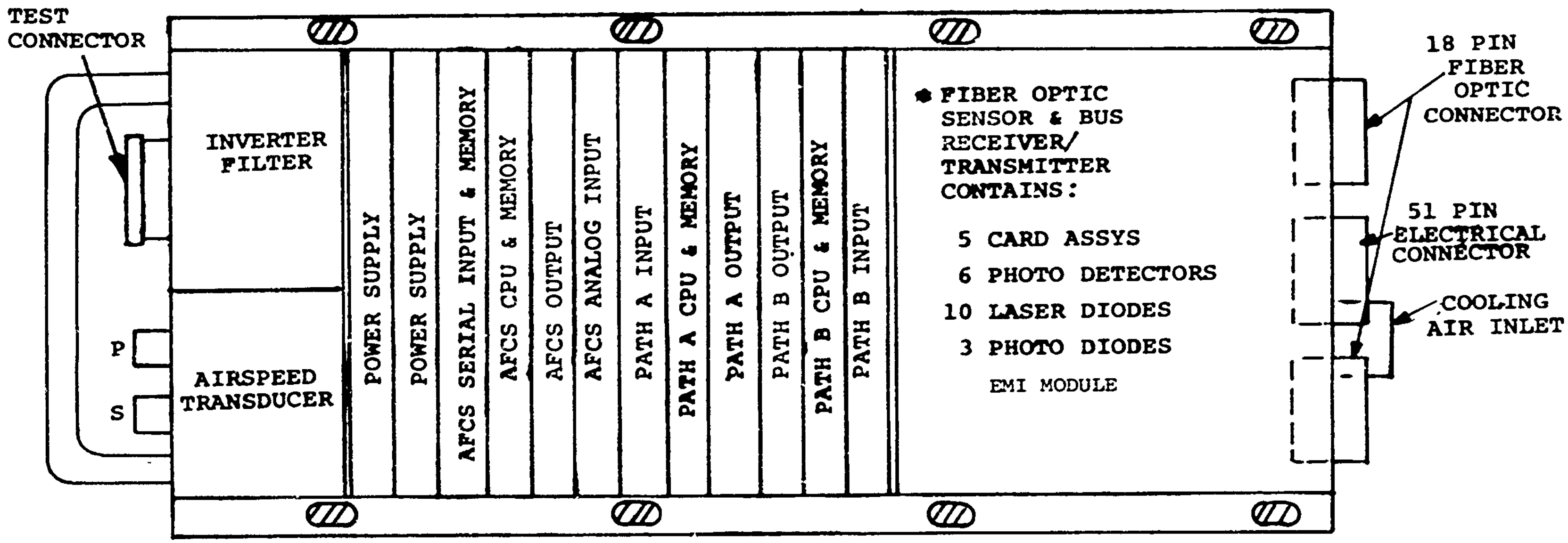
1. Two additional plug-in cards (total of 14)
2. Smaller fiber optic module
3. Larger EMI module
4. Large number of connectors on front of unit.

Figure 15 shows the flight control processor used with the dual mechanical primary flight control system. This unit is identical to baseline except for:

1. Two fewer plug-in cards (total of 10)
2. Smaller fiber-optic module
3. Large EMI module
4. No PFCS is provided.

- <u>Sensor Multiplex/Test Interface Unit</u>

Figure 16 shows the information flow diagram for the sensor multiplex/test interface unit, which is used with all three system options to provide an electrical interface with several



NOTES:
1. SIZE = 7.5 x 18.0 x 7.62 IN. (BASED ON 3/4 ATR CASE)
2. WEIGHT = 25.7 LB
3. ELECTRICAL POWER: 28 VDC AT 159 W
*4. FIBER OPTIC MODULE USES UPPER 2/3 OF AREA. EMI MODULE USE LOWER 1/3.
5. CONNECTORS - REAR MOUNT MIL-C-83733 (REFERENCE 9)

Figure 13. Flight Control Processor - Baseline Flight Control System.

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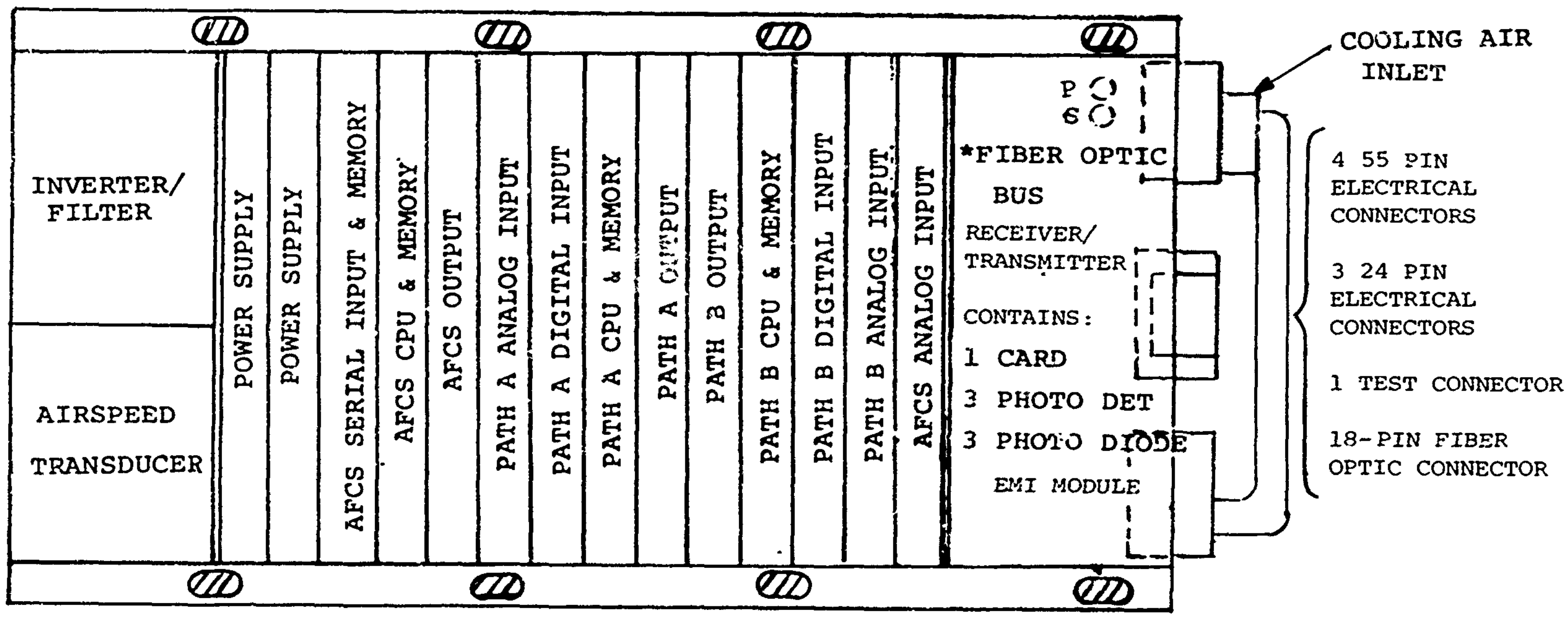
9. Military Specification, MIL-L-83733/3A, CONNECTOR, ELECTRICAL, RECEPTACLE, MINIATURE, RECTANGULAR TYPE, BACK TO PANEL, WITH GUIDE SOCKETS, ENVIRONMENT RESISTING, ZOO DEG. C, Department of Defense, Washington, D.C., 19 April 1976.



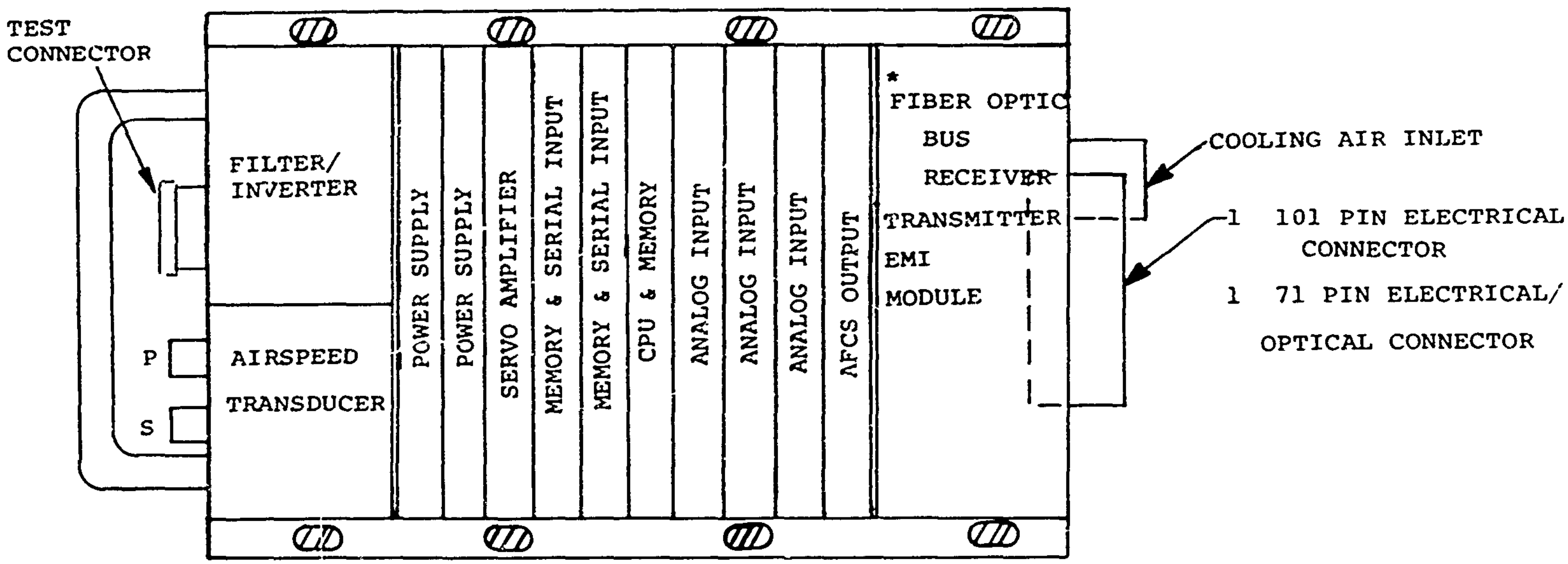
NOTES:

1. SIZE = 7.5 x 16.0 x 7.62 IN
   (BASED ON 3/4 ATR CASE)
2. WEIGHT = 22.9 LB
3. ELECTRICAL POWER: 28 VDC AT 150 W

*4. FIBER OPTIC MODULE USES UPPER 1/3 OF AREA AND EMI MODULE USES REMAINDER

5. CONNECTIONS ON FRONT OF UNIT MIL-C-83723-III THREADED

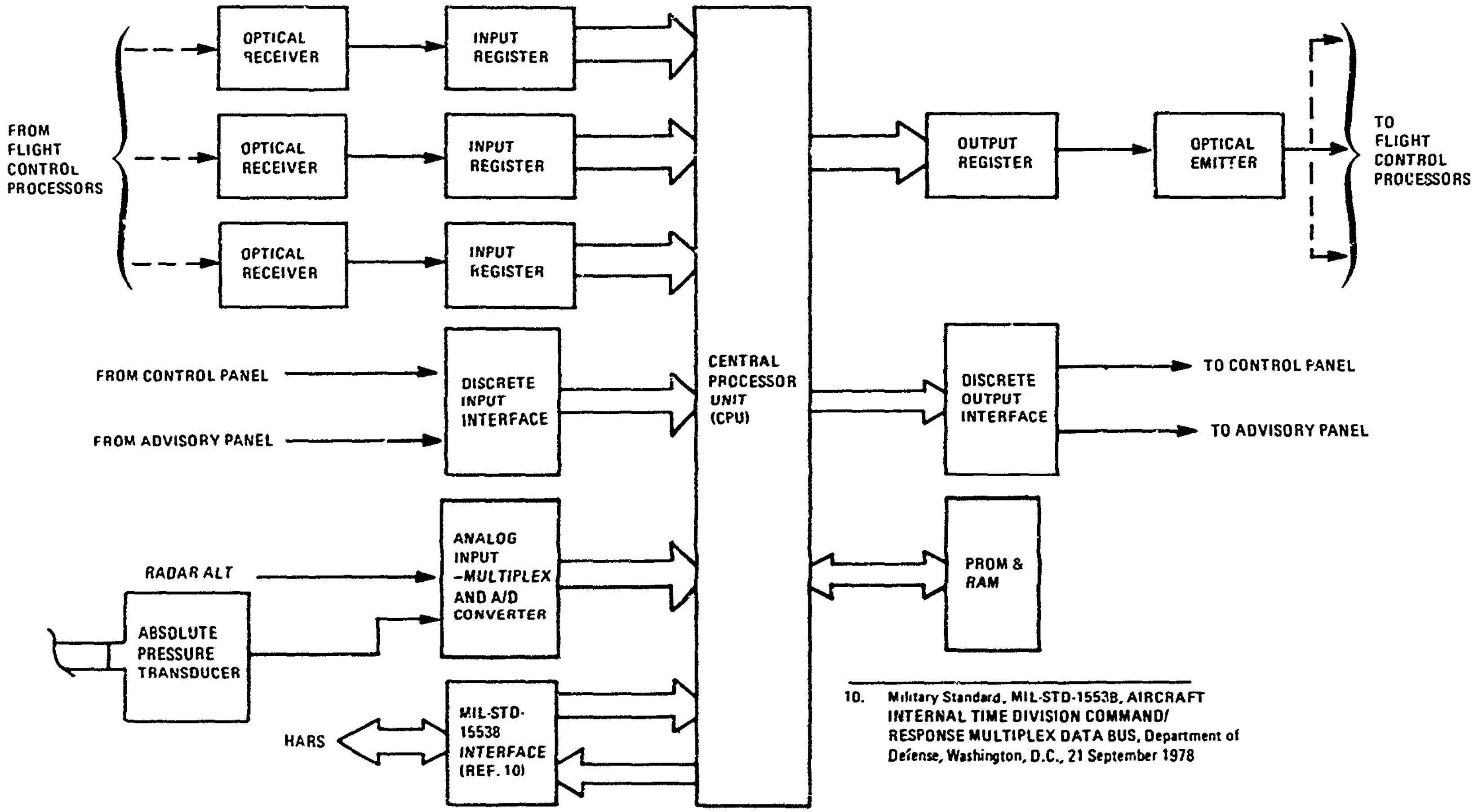
Figure 14. Flight Control Processor - Alternate Flight Control System.



NOTES:

1. SIZE = 7.5 x 13.3 x 7.62 IN. (BASED ON 3/4 ATR CASE)
2. WEIGHT = 19.0 LB
3. ELECTRICAL POWER: 28 VDC AT 98 W
*4. FIBER OPTIC MODULE USES UPPER 1/3 OF AREA AND EMI MODULE USES REMAINDER
5. CONNECTORS - REAR MOUNT MIL-C-83733/3A

Figure 15. Flight Control Processor - Dual Mechanical Flight Control System.



10. Military Standard, MIL-STD-1553B, AIRCRAFT INTERNAL TIME DIVISION COMMAND/RESPONSE MULTIPLEX DATA BUS, Department of Defense, Washington, D.C., 21 September 1978

Figure 16. Sensor Multiplex/Test Interface Unit - Information Flow Diagram.

single-channel devices and an optical interface with each of the three flight control processor units. An Intel 8086 processor is included to control the input/output functions.

Figure 17 shows the sensor multiplex/test interface unit. A 3/4 ATR chassis provides the enclosure for:

1. Test interface/display
2. Seven plug-in card assemblies
3. Absolute pressure transducer barometric altitude
4. Inverter/filter assembly
5. EMI module
6. Fiber-optic bus transmitter/receiver module.

The assemblies are similar to those used in the FCP.

- Control Panel

A control panel provides for primary channel reset, AFCS reset, and selection of mission-oriented modes (i.e., hover hold, altitude hold).

Software

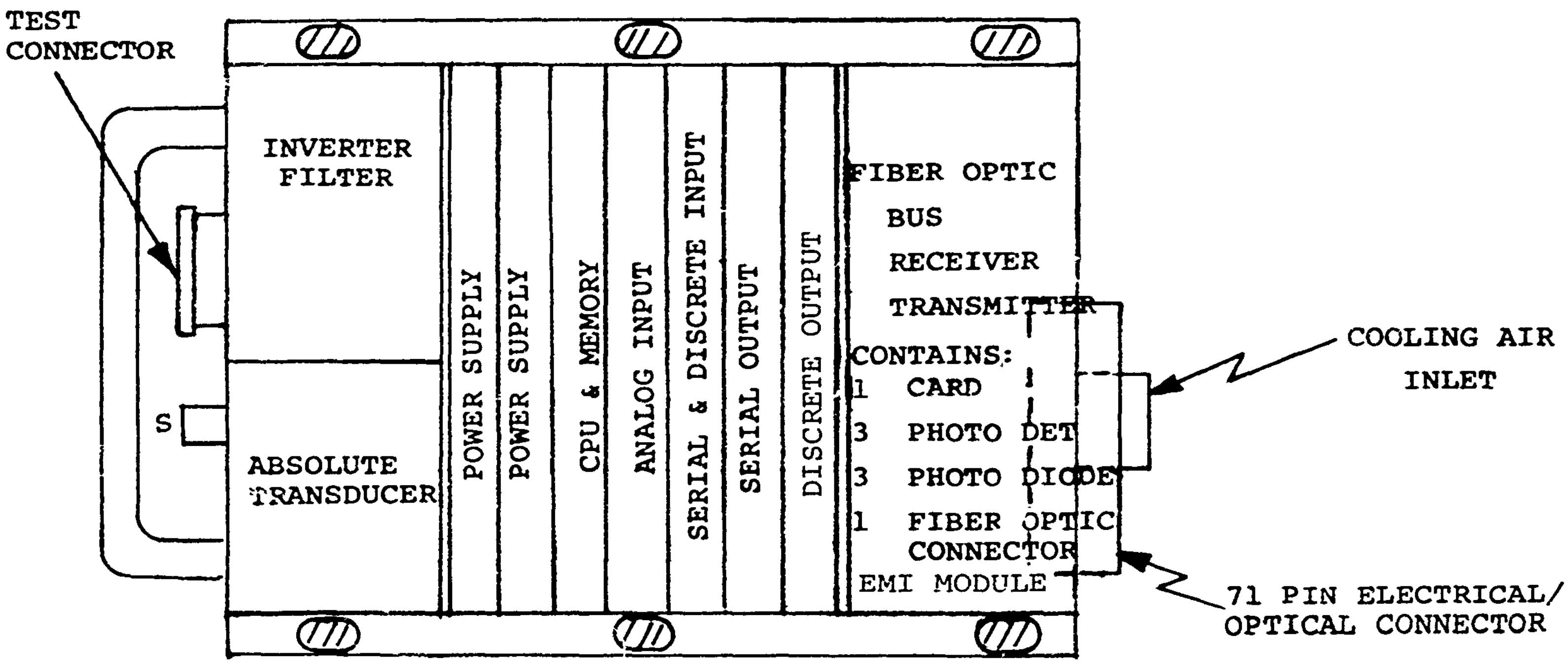
Software loads were estimated for the baseline (FBO) PFCS processor and the AFCS processor, as follows:

Function	PFCS Processor	AFCS Processor
Percent Time Loading	70%	85%
Memory Requirement	2.8K	6K
Control Law Sample Rate	40 per second (160 per second fiber optic sensor preprocessing and servo loop closure)	40 per second

The PFCS time loading is reduced for the alternate approach (fly-by-wire), since the servo loops would be closed directly with the analog signals.

Failure Monitoring/Built-In Test Equipment (BITE)

Functions are included in each system to allow detection of more than 99 percent of all faults that occur and to allow isolation of at least 90 percent of all faults to the offending LRU. The fault detection and isolation encompasses the flight control processor units, sensor multiplex/test interface unit, and external sensors and actuators.



NOTES:

1. SIZE = 7.5 x 11.0 x 7.62 IN.
   (BASED ON 3/4 ATR CASE)
2. WEIGHT = 15.8 LB
3. ELECTRICAL POWER: 28 VDC AT 82 W
4. FIBER OPTIC MODULE USES UPPER 2/3 OF AREA

   EMI MODULE USES LOWER 1/3
5. CONNECTORS REAR MOUNT MIL-C-83733/3A

Figure 17. Sensor Multiplex/Test Interface Unit - (All Candidates).

ROTOR CONTROL ACTUATOR

Functional Description

Figure 18 is a block diagram of the baseline rotor control actuator and its interface with the flight control processor. This diagram shows how one path of a typical processor controls the actuator. Note that all three processors interface with the EHV and position transducers. Under normal conditions the spring detents are rigid. This means that each channel receives a dual control stage position signal. This allows for mechanical failure of the drive to one transducer without loss of this "inner loop" feedback.

Figure 19 is a schematic of the baseline rotor actuator. The two-stage design incorporates a duplex control stage driving a duplex power stage. Current commands from each of the flight control processors are magnetically summed at each control stage. The single stage jet-pipe type electrohydraulic valves (EHV) control the pressure and fluid flow to the unbalanced control stage pistons. System pressure acts on the small side of the piston. This pressure is balanced by the EHV output pressure. The area ratio is 2:1. Use of the unbalanced design is key to the detection of certain passive failures of the EHV and piston (see Redundancy Management section).

The position of control stage pistons is measured by triplex optical displacement transducers. These units provide for inner loop feedback and for monitoring of control stage performance.

The control stage pistons position the power stage control valve through spring detent mechanisms. These mechanisms allow control of the power stage valve in the event of a control stage jam. They also allow detection of a failed control stage (see Redundancy Management). The power stage valve is an anti-jam type design. The valve meters flow to the power stage to produce power stage piston velocity in proportion to its displacement.

The duplex power stage is made up of three pistons. The inner piston constitutes one system; the outer two pistons (which are hydraulically and mechanically in parallel) constitute the second system. This approach will eliminate bending of the pistons due to pressure mismatch in normal or single system operation.

Optical position transducers that are concentric with each of the three positions provide the triplex position feedback necessary to close the outer power stage servo loop. The actuator contains a hydraulically operated valve to shut off and bypass a failed actuator section.

Hardware Details

Figure 20 is a schematic of the rotor actuator as proposed by Bendix Electrodynamics Division. Figure 21 shows the actuator assembly and details of the power stage piston. Figure 22 shows details of the pressure operated shutoff valve, control stage piston assembly, and power stage control valve.

- Manifold

The control stage piston assembly controls the dual tandem power stage control valve through an equal leverage linkage. The power stage control valve is a dual concentric spool configuration.

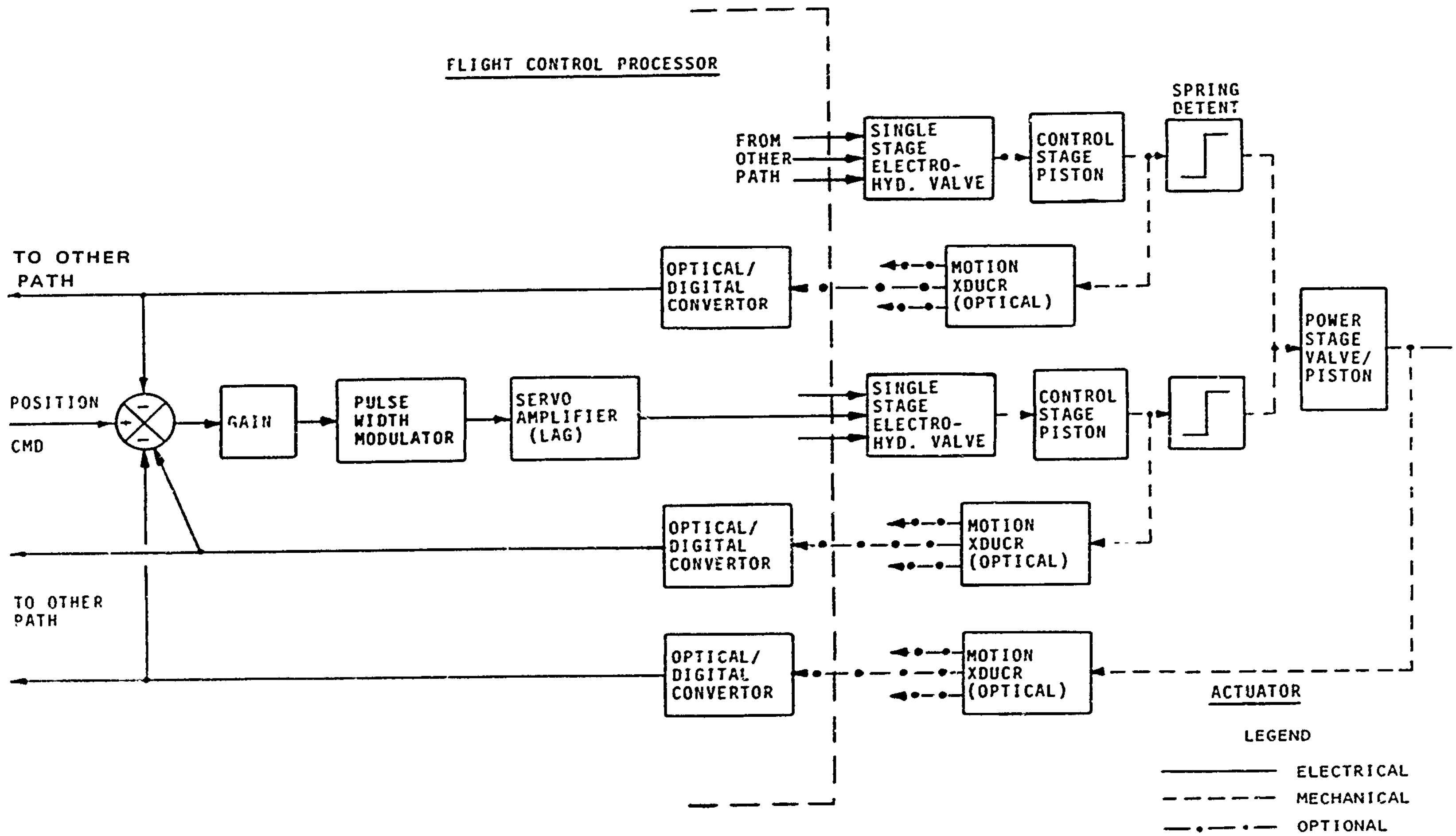
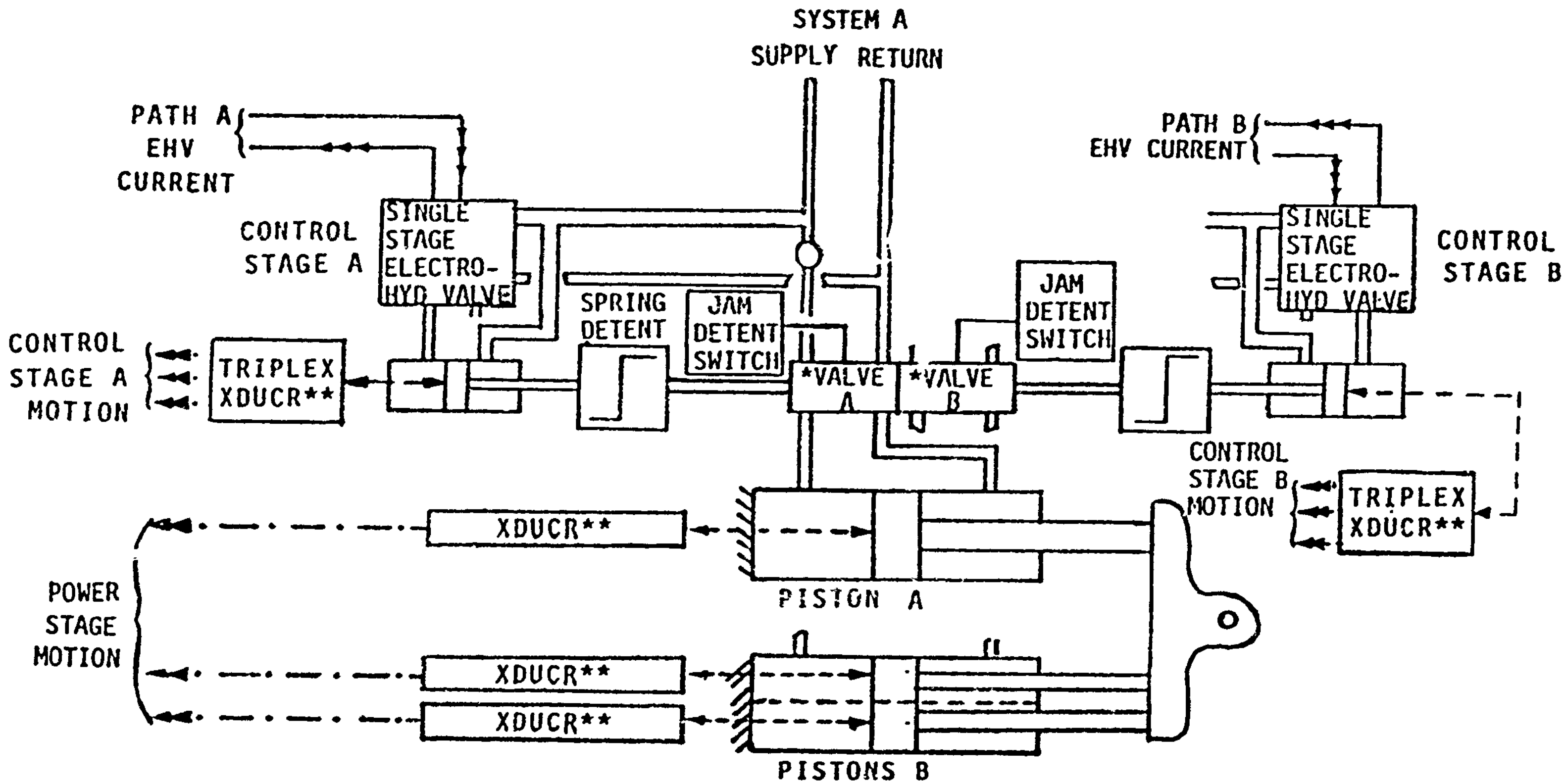


Figure 18. Baseline Rctor Control Actuator - Servo Loop.



**OPTICAL OR ELECTRICAL POSITION TRANSDUCER

LEGEND:
——— ELECTRICAL
– – – – MECHANICAL
– · – · – OPTICAL OR ELECTRICAL

*EACH VALVE SECTION INCLUDES ANTI-JAM VALVE WITH PROVISIONS FOR SUPPLY SHUTOFF AND CYLINDER BYPASS

Figure 19. Rotor Control Actuator.

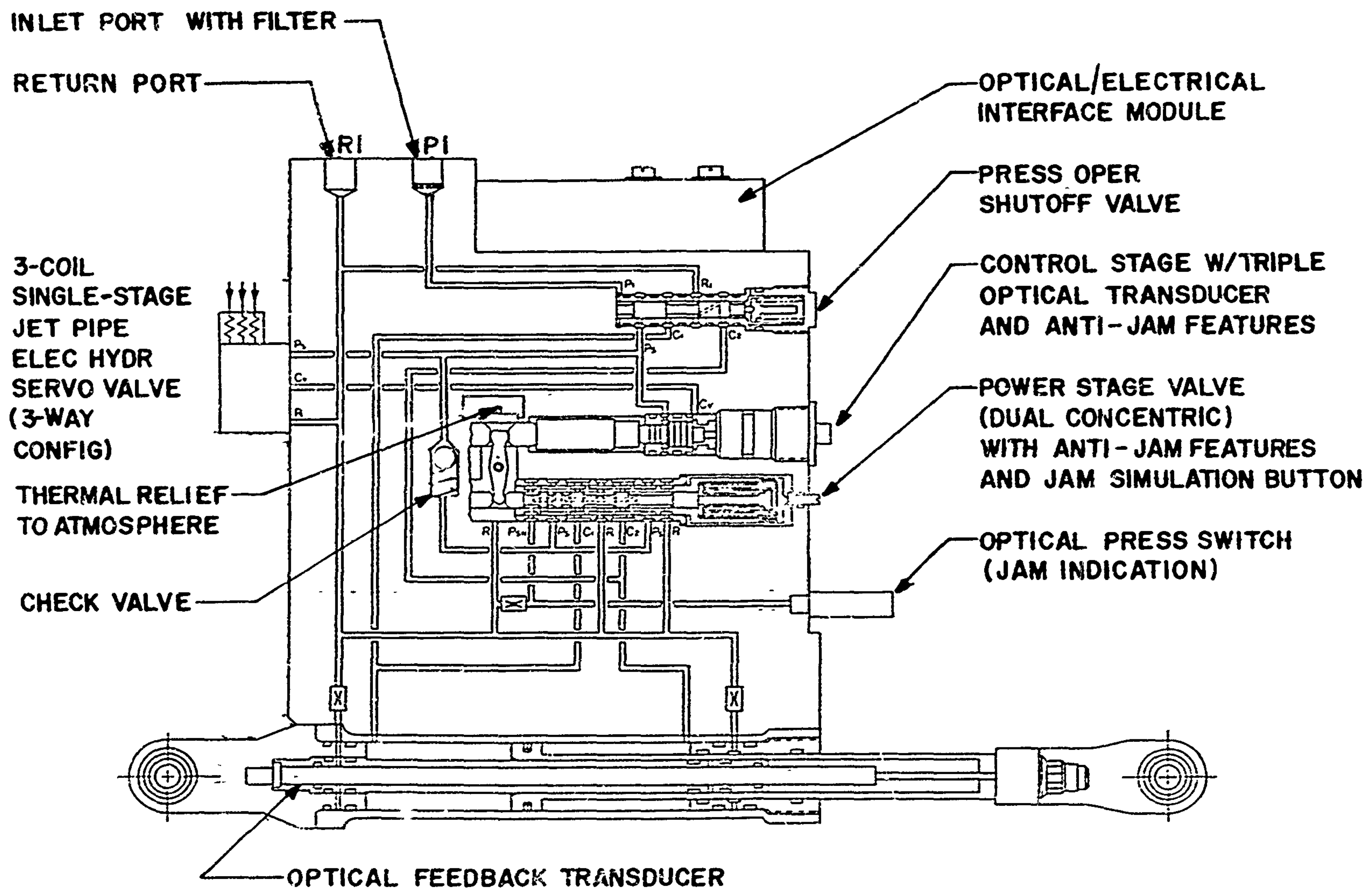


Figure 20. Main Rotor Actuator Schematic (Single System Shown).

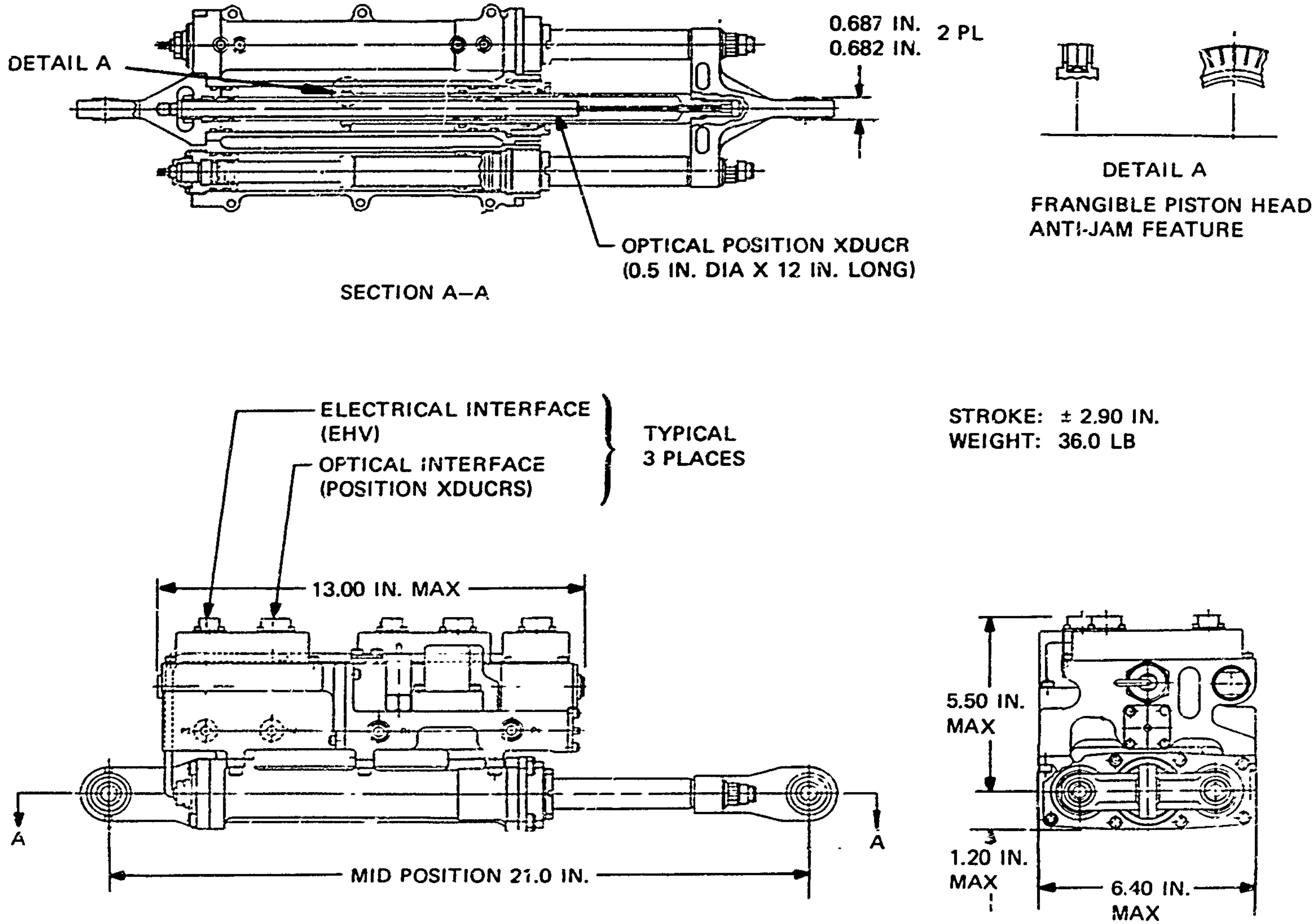
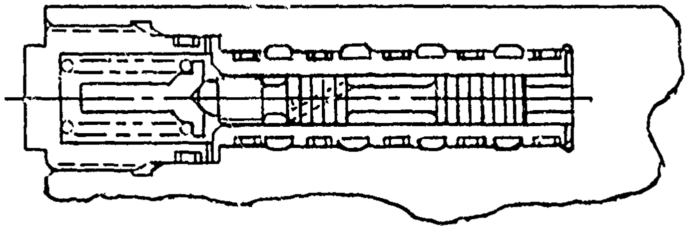
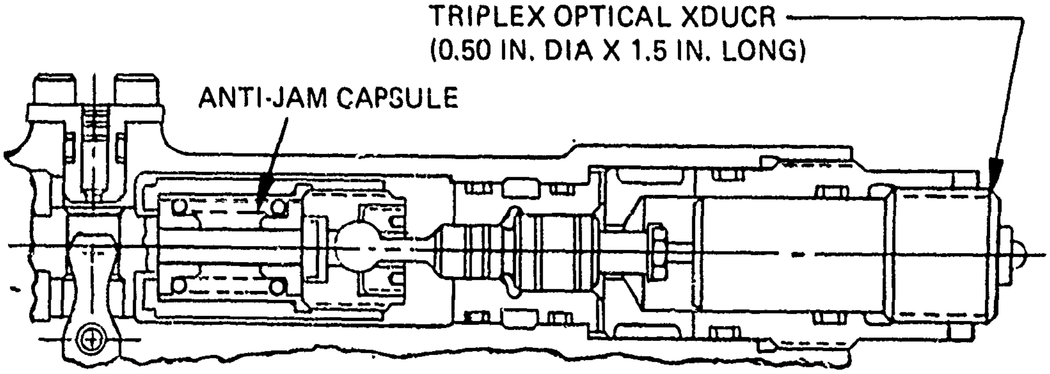


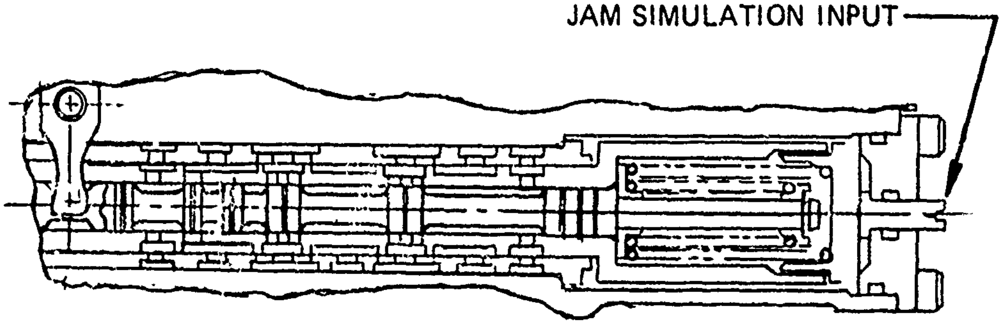
Figure 21. Main Rotor Control Actuatcr - Baseline Flight Control System.



(A) PRESSURE OPERATED SHUTOFF VALVE



(B) CONTROL STAGE PISTON ASSEMBLY



(C) POWER STAGE CONTROL VALVE

Figure 22. Rotor Actuator Details - Baseline Flight Control System.

The primary control is accomplished by metering around the outer spool; anti-jamming control is provided by the inner spool, which is referenced to a caged spring assembly. The cylinder lands are the metering lands. System pressure is applied at the outer lands with system return at the inner lands. A pressure switch is incorporated as an indication of jamming. In case of a jam, system pressure will be connected to the adjacent cavity and will trigger the pressure switch. The valve design also includes a manual jamming "checkout". The pressure-operated shutoff valve blocks inlet pressure and connects the cylinder lines to return if system pressure is below 200 psi. In the event of control stage failure or power stage valve jam, pressure to the actuator will be shut off using the system shutoff valve located in the system manifold (see Redundancy Management).

- Power Stage Cylinder

The power stage cylinder is a dual parallel design. The two outer barrels are controlled by one hydraulic system power control valve and the inner barrel is controlled by the other hydraulic system power control valve. The actuator output of the three barrels is tied together by a single rod end and also with one clevis, which is sized to withstand a 7.62 mm round and to limit growth of fatigue cracks. All piston rods are initially torqued to the stall force level. The barrels are a rip-stop design with each barrel manufactured separately. Ballistic anti-jamming features include frangible piston heads and protrusion-tolerant glands. A bronze-Teflon liner of sufficient length is part of the aluminum gland in accordance with the required L/D ratio to minimize cocking. The feedback-optical position transducers are mounted within the piston rod for compact design and environmental protection. All barrels are of corrosion resistant steel, heat treated to 150 ksi. The control stage piston manifold is a two-piece brazed design to meet the rip-stop requirements and is manufactured from corrosion resistant steel. The power stage control valve sleeve is press (shrink) fitted to reduce the overall package length.

- Anti-Jam Provisions

Two anti-jam provisions have been included in the design: anti-jamming series springs in the control stage piston assembly and anti-jamming springs in the power stage control valve assembly. The control stage piston jamming springs will be activated if the jamming force exceeds the equivalent of 110 pounds. The jamming springs have a very high rate to minimize impact on system dynamic performance. A hydraulic jamming detent could be used in lieu of these coil springs, however it is more complex, requiring a reducing valve. It could be studied more fully in a detail design phase. The power stage control valve anti-jamming springs will be activated if the jamming force exceeds the equivalent of 40 lb at the power stage control valve centerline. In this case, if jamming occurs, the cylinder lines are connected to return and the system pressure lines are connected to the jam indication pressure switch, which actuates the external shutoff valve and sets a warning on the caution/advisory panel.

Dual Mechanical Flight Control Boost Actuator

Figure 23 is a block diagram to the integrated boost/AFCS actuator designed for the mechanical control option. This unit accepts inputs from pilot and copilot via separate inputs. Anti-jamming is achieved by a spring capsule in the valve input linkage. The AFCS actuator is dual

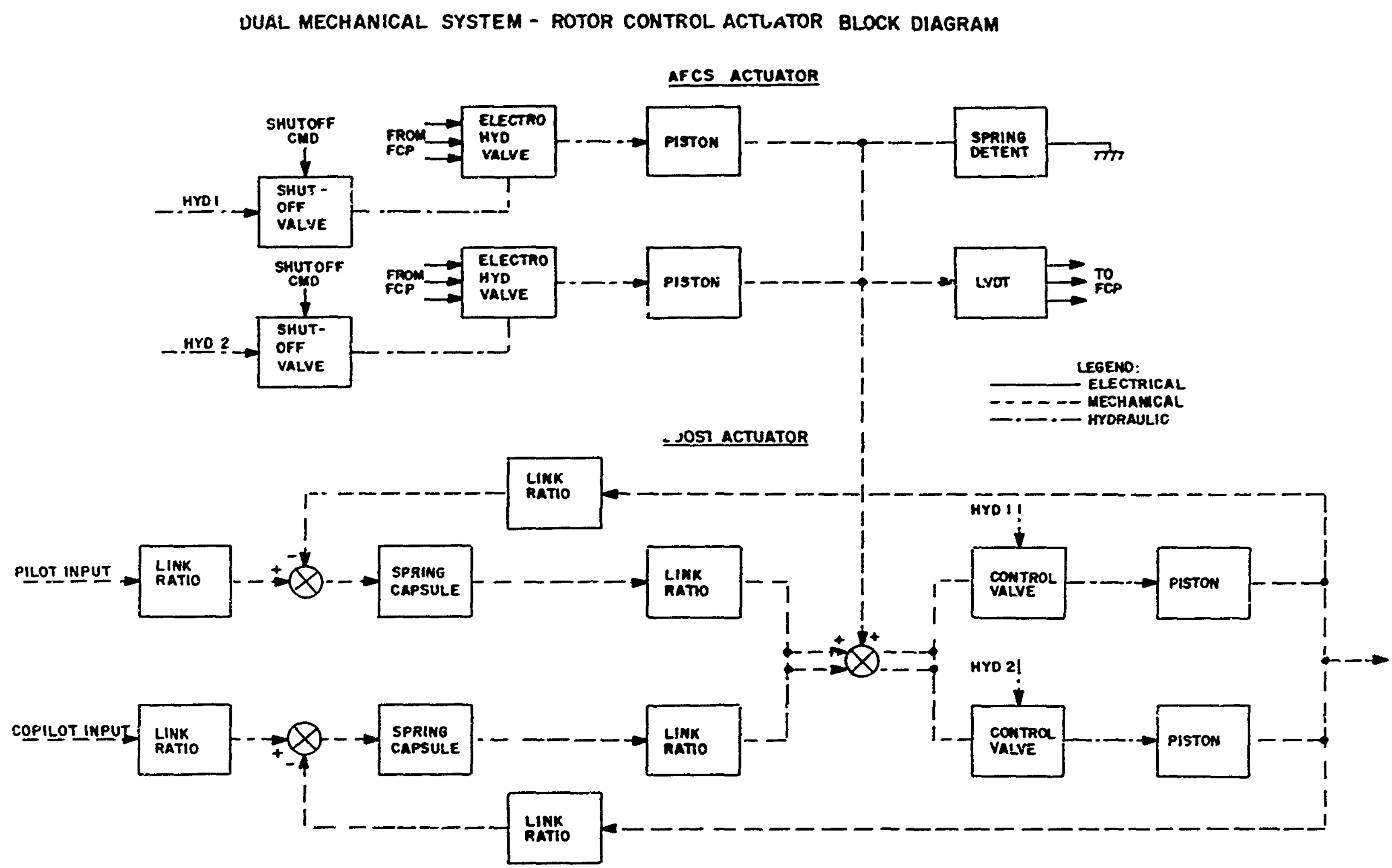


Figure 23. Dual Mechanical Flight Control System - Rotor Control Actuator Block Diagram.

with triplex electrical control. It interfaces with the control valve of the boost section. When both sections of the AFCS are hydraulically shut off, the spring detent maintains the actuator fixed at center.

Figures 24 and 25 show the Bendix Electrodynamics Division design for the actuator. This actuator is similar to the baseline unit except that the control stage (AFCS actuator) is designed for 15-percent authority. Linkages are based on 50-percent gain with out-of-phase polarity (input/output). Anti-jamming provisions are provided at the control valve. The linkage is dualized, with a dual load path at the control valve interface. The external spring protects against jams in the linkage up to the capsule. The input crank to the valve would be covered to prevent trapping foreign objects between the crank and the housing.

REDUNDANCY MANAGEMENT

To achieve the specified failure reliability goal (i.e., no more than one loss of aircraft due to flight controls in 10 million flight hours – based on a 1-hour mission), the baseline primary system was configured with the following levels of redundancy:

1. Sensors (optical path), PFCS signal processing, and PFCS electro-optical interconnect lines: Dual-fail operative.
2. Hydromechanical portions of the actuators and hydraulic lines to actuators: Single-fail operative.
3. Electrical power supply: Dual-fail operative.
4. Hydraulic power supply: Dual with switched limited capacity third channel for backup.
5. Mechanical portions of sensors: Single-fail operative checked in a background inter-channel comparison.

To enhance mission capability the AFCS was configured single-fail operative for basic stabilization and fail-safe for altitude and hover hold functions. The following discussion gives details of the redundancy management scheme, starting at the system level.

Overall System

The in-line (self) monitored concept of redundancy management used for the primary system involves the use of two identical signal paths in each channel between the cockpit controls and the actuator input (Figure 26). If a discrepancy occurs that is greater than a preestablished tolerance level, that channel is considered to have failed and is shut down. The electrohydraulic actuators have dual hydraulic sections and triplex electrical sections. Thus, in comparison to the mechanical approach, the dual mechanical linkage is replaced with a triplex electro-optical link, while a dual hydraulic power section used in the mechanical approach is maintained in the electro-optical mechanization. Each channel of the electro-optical link is powered by an independent electrical supply. Tracking of three channels is maintained by control of overall gain tolerances which is enhanced by use of digital transducers and processing. Channel inputs are summed magnetically in the electro-hydraulic valves of the actuator. Inherent fail safety

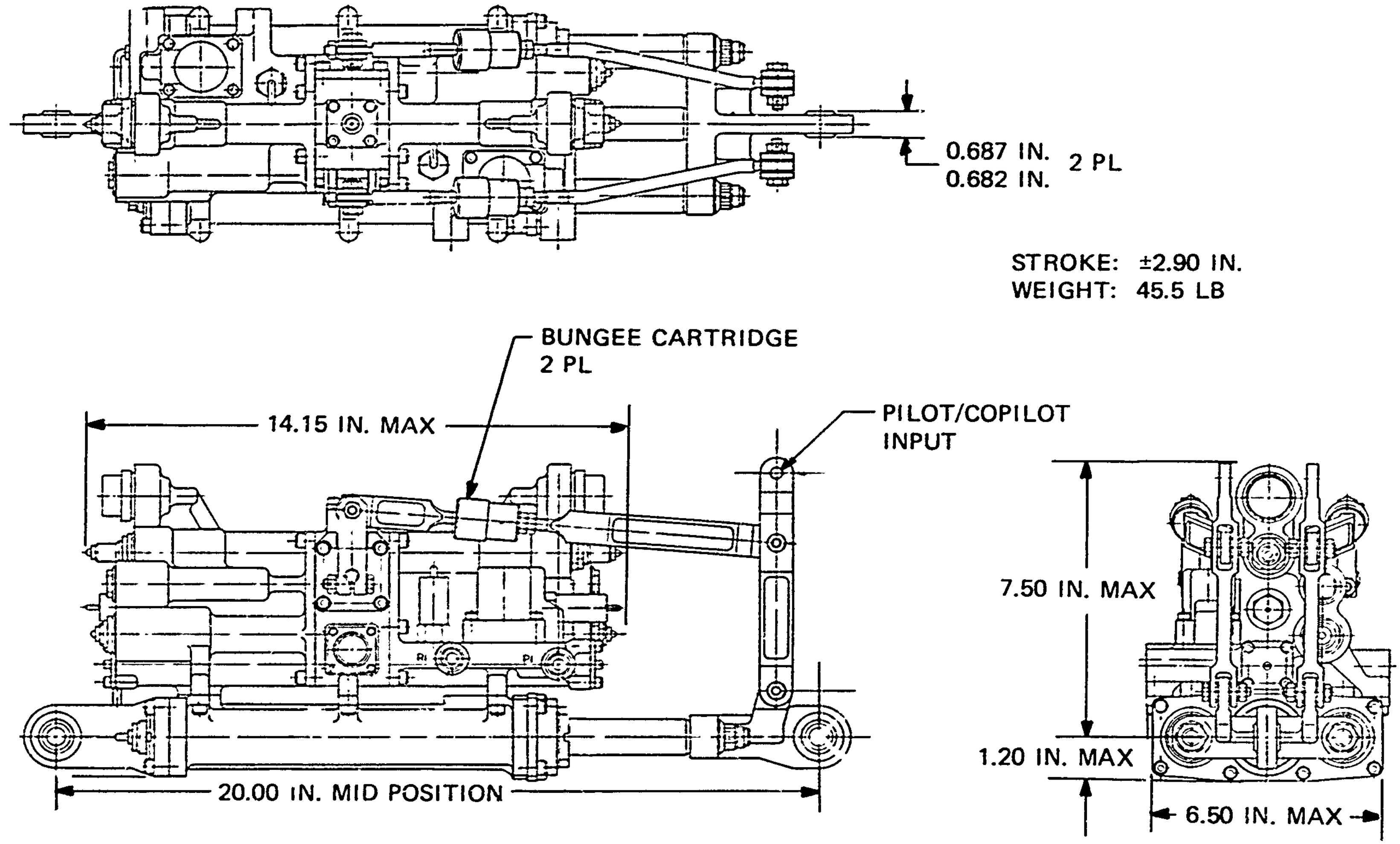


Figure 24. Main Rotor Control Actuator - Dual Mechanical Flight Control System.

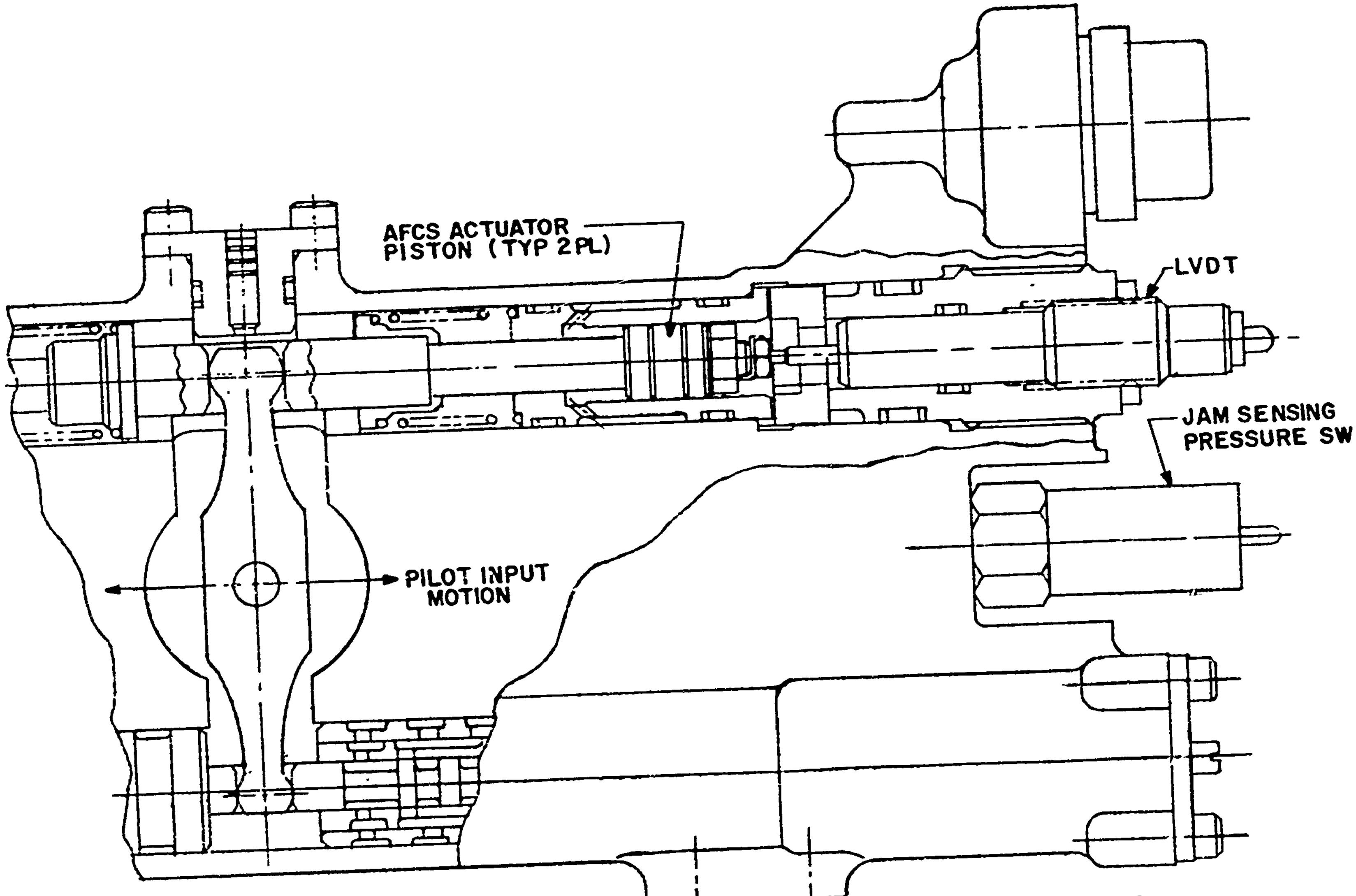


Figure 25. Details of Main Rotor Control Actuator - Dual Mechanical Flight Control System.

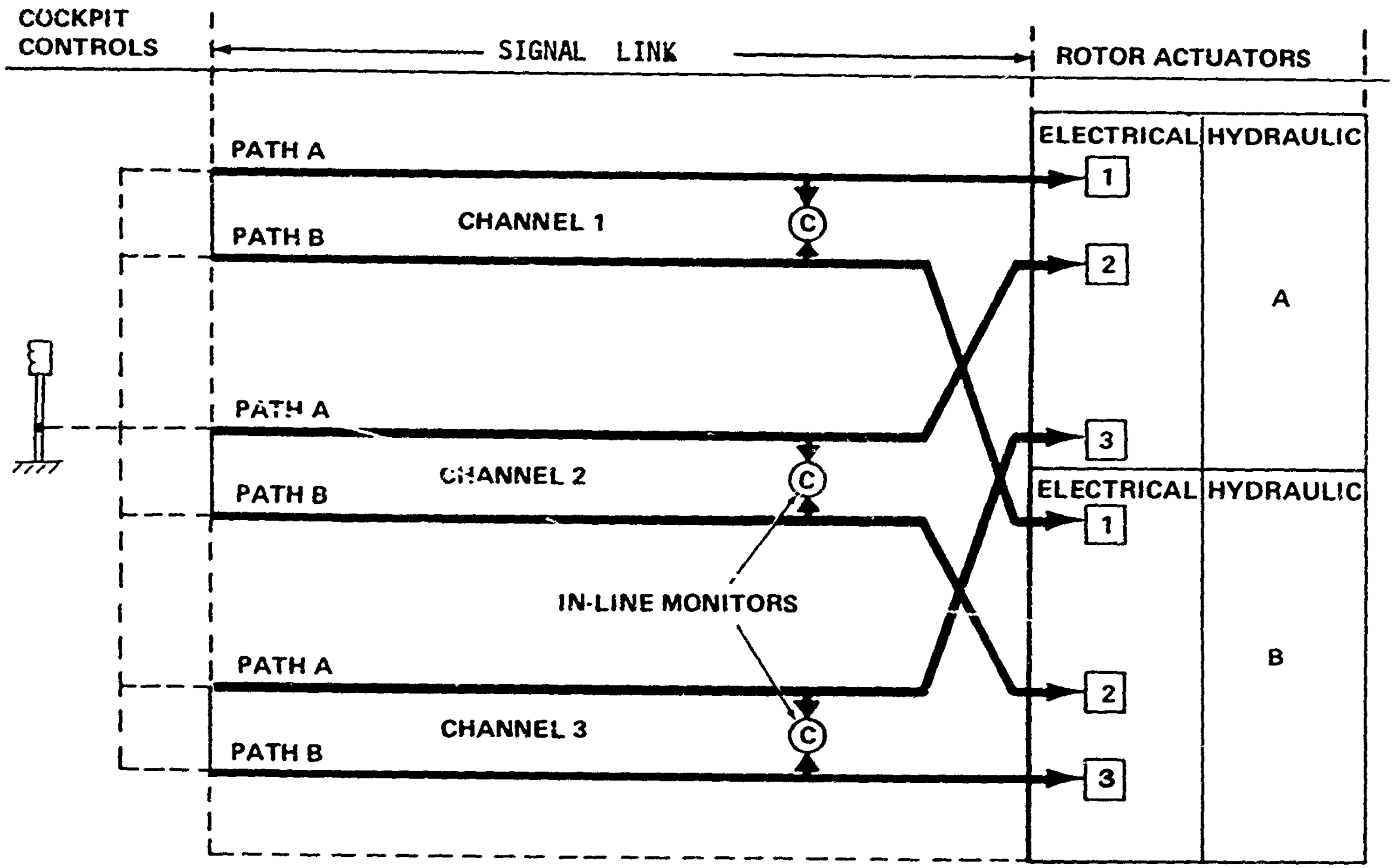


Figure 26. Redundancy Management Concept.

without time-critical switching is maintained for first failures by use of magnetic summing. The following sections discuss detailed methods used in each major portion of the system.

Optical Sensors

Each LRU (which contains multiple sensors) is excited by a single emitter. Proper performance of the emitter and optical cabling is checked by monitoring for a parity bit which will be returned from each sensor when the circuit is complete. The parity bit is checked during each excitation cycle. If the bit is lost, the channel's current input to the actuators is removed.

In the case of the sensor in a pilot control, a channel reset is allowed if the controller with the failed sensor has been disabled by the controller cutout function (see the Cockpit Controls section).

Failure of a portion of the fiber-optic sensor (i.e., loss of one or more bits) is detected by the downstream comparison of actuator input current.

For cockpit control and main actuator feedback transducers, failure of the mechanical drive to an individual sensor (a remote occurrence) is detected by cross-channel comparison of the three channels. This test is run as a startup BITE check and possibly could be run in flight on a low cycle rate. If a miscompare is noted, the pilot is notified but no channel shutdown is allowed.

Mechanical failure of the control stage position transducers are detected by comparison of valve current and actuator position (see Control Stage Hydromechanical Failures, below).

Cockpit Controls

Jams, opens, and other gross mechanical failures of the cockpit controllers are removed by switching out the failed unit. This mechanization also allows for removal of a hardover, which could occur if the pilot was wounded. The mechanization of this function is shown in Appendix A (Figures A-7, A-8, and A-9). If a pilot detects a hardover input, he counteracts it with his controller and then presses the disconnect button on his control. After this, the other pilot's controller output is ramped to zero via a synchronizer at a rate that allows the overriding pilot to follow, thus maintaining control of the vehicle. When the failed controller synchronizer output is zero, it is latched out until the controller output and synchronizer output are both zero and the system reset is actuated. The system logic prevents both controllers from being disengaged at the same time.

An alternate scheme (which was considered to cover the pilot force flight case) was the provision of an extended dynamic range for the control (i.e., large enough to permit overpowering a wounded pilot). This scheme was dropped because it does not cover the case of controller mechanical failure and also would require flight with excessive stick forces.

Signal Path Monitor

Comparison of calculated command input to the EHV is used to detect failures in the path A/B processing from sensor output to actuator current return (i.e., actuator wiring and EHV coil).

Figure 27 shows the mechanization of the signal path comparator. This comparison is made in two steps. The outputs of Path A and B digital-to-analog converters are compared to detect failures in the sensors and electronics. After comparison, the signals are averaged in the servo-amplifiers. This removes any accumulated error due to tolerance stackup from the next stage of comparison.

Electrohydraulic valve currents of each actuator section are compared with a third servo amplifier, which drives an EHV torque motor model. This comparison allows detection of failures in the servo amplifier, actuator wiring, or EHV torque motor coil. The model EHV torque motor is used to prevent shutdown of both actuator sections, should one EHV suffer a ballistic hit, by providing an independent third voting reference.

Control Stage Hydromechanical Failures

Failures downstream of the EHV coil are detected by comparison of valve current with the control stage piston position. The unbalanced control stage design eliminates the passive failure modes, which can require a special test excitation to assure detection. If we first consider a piston-balanced design, we note that the normal condition of the actuator at rest is zero current and zero differential pressure. Any failure that does not disturb the differential pressure will not be detected. Piston seal failures and blockage of the inlet jet pipe are in this category. With an unbalanced design, a unbalanced pressure must be maintained to hold the piston in equilibrium. This design does not have passive failure modes.

Detection of all failures in the control stage is made by comparison of piston position and input current. When a hardover failure occurs, it is opposed by the other channel, resulting in collapse of the anti-jam override spring capsules. Under this condition, control of the power stage is lost until the failed actuator is shut off. With both spring capsules collapsed, there will be a slow rate displacement of the power stage valve because power stage valve offset is limited to that allowed by neutral rigging tolerances.

The current versus displacement comparators (Figure 27) for all actuators are "ORED" to control the hydraulic system shutoff valve (Figure 28). When the system is reset the contacts of relays K_A and K_B are held open. Both path processors monitor for failures and control individual shutoff valves. When all three channels agree on a failure, power is applied to the valves. When either valve is energized and pressure is shut down, control of the actuator is restored The delay is of the order of 50 milliseconds. Since the power stage control valve is centered, there will only be a small power stage motion.

Separate 28 VDC feeds are provided for each valve. The pressure switches in the return side of the shutoff valves prevent shutdown of both hydraulic supplies. The test switch is used for pre flight check of the system.

Failures that result in a control stage jam, particularly near center, will present a slightly different detection problem. In this case control is not lost. The output of the current-versus-displacement comparator will be smaller and cyclical as the non-failed side moves the valve via the anti-jam capsule. For this case, the comparator output will be accumulated over a period of time after which hydraulics would be shut down and pilot notified (this feature is not

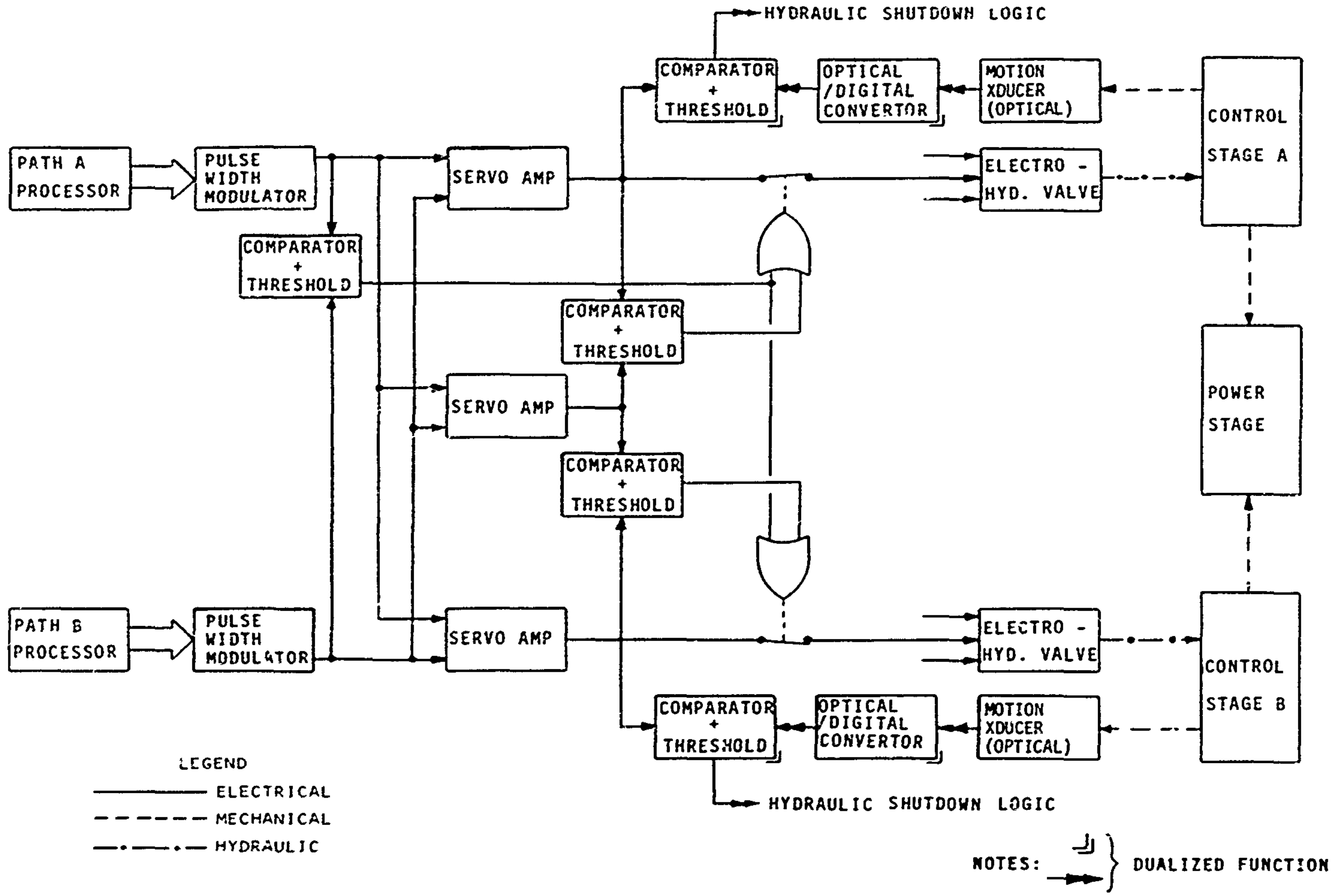


Figure 27. Baseline Rotor Actuator Redundancy Management.

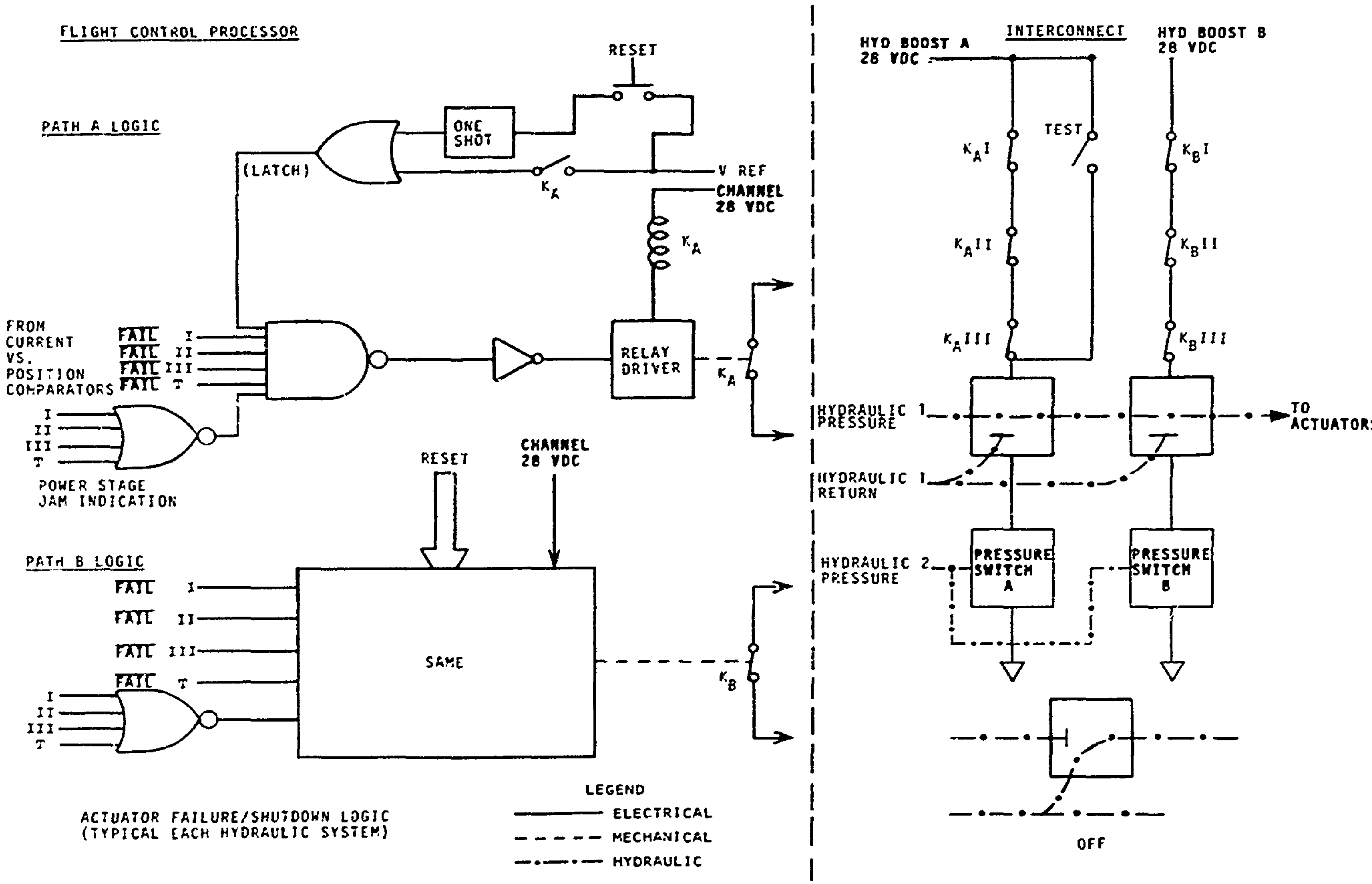


Figure 28. Actuator Failure/Shutdown Logic (Typical Each Hydraulic System).

illustrated). When power to a channel is lost, its relays are closed. This enables hydraulic shutdown by the two remaining channels if a subsequent control stage failure is detected.

Power Stage Failures

The linkage from the control stage to the power stage is of a dual load path design. As noted previously, the power stage control valve is of an anti-jam type with pressure sensing switches in each system to detect operation of the jam override mechanism. Operation of these switches is also "ORED" in the hydraulic system shutoff valve (see Figure 28). On shutdown, the pressure-operated shutoff/bypass valve built into the actuator provides bypass around the failed power stage.

Clearance of the jams of the power stage piston is accomplished by frangible piston heads and glands, which break out to allow operation after projectile penetration.

Opens in the power stage pistons are handled by the duality of the design. Attachments are designed to limit growth of fatigue cracks and to survive ballistic damage. Under certain conditions of mounting, these attachment points could be made redundant.

In general, detection of jam and open failures of the power stage would be by visual inspection. Failures resulting in fluid loss would be detected via hydraulic system instrumentation.

The design is not protected for ballistic damage to the power stage control valve or its drive linkage. This vulnerable area can be protected by positioning it next to heavy structure, by armor plating, or acceptance of this small vulnerable area. The Boeing survivability/vulnerability analysis includes this portion of the actuator in the single hit vulnerable area.

Automatic Flight Control System (AFCS)

Redundancy management for the AFCS consists of voting and failure detection of the sensor inputs, voting and failure detection of computer AFCS inputs to the PFCS, and velocity and rate limiting of the selected AFCS inputs to the PFCS.

- Sensor Inputs

Details on AFCS sensor inputs are shown in Appendix A. They consist of triplex stabilization sensor signals and the pilot's primary control outputs (see Figure 12). Within a channel, these signals are sampled serially and passed to adjacent channels via fiber-optic links. Adjacent channel signals are sent to this channel's AFCS via fiber-optic links and the dualized PFCS processors. This method provides separate processing of the adjacent channel signals (this is particularly important for the computed results, which also take this path).

The three sets of sensor data are voted to select the one to be used in the AFCS computation. The voter also provides failure detection to reject a failed sensor. Voter processing provides a frequency/amplitude sensitive threshold which allows a larger disparity to exist if it occurs over a period of time, but votes out a smaller, faster building, disparity more quickly. This technique was used in the HLH AFCS.

Nonredundant sensor inputs come to the FCP via a fiber-optic input from the Sensor Multiplex/ Test Interface Unit.

Details on nonredundant sensor monitoring are given in Appendix A. For example, the performance of the barometric or radar altimeter signal is monitored by comparing it with the vertical position reference from the heading and attitude reference system (HARS). The HARS and doppler velocity outputs are compared to detect failures in these devices. Where a nonredundant sensor failure is detected, the mode using it (i.e., Altitude or Hover Hold) is cancelled with a minimum transient disturbance and the system reverts to attitude hold.

- Computed Results

The computed results of all three channel AFCS processors are brought serially to each PFCS processor where they are voted. The selected signals are then passed through authority and rate limiting functions (see Appendix A, Figure A-20) and summed with pilot's commands. The axis summation signals are then mixed and sent to the rotor actuator. These voters and limits are part of the path processor. Failures of these functions are detected at the actuator.

POWER SUPPLIES

Electrical Power

Each channel is supplied with 28 VDC as the primary source of electrical power. A second 28 VDC input to each processor is provided for the non-flight safety critical horizontal stabilizer actuator. Figure 29 shows the scheme used to supply the primary 28 VDC. Three in-flight sources of power are used – the two aircraft generators and a third small permanent magnet generator (which is added for FBO/FBW). Loss of various supplies results in fall back to alternate supplies. This scheme gives an extremely low probability of in-flight electrical power loss. For ground operation, power is supplied from the APU or an external source.

Hydraulic Power

The hydraulic power supply for all three candidates is essentially the same (Figure 30). The only differences for FBO/FBW versions are (1) a shutoff is provided at the ground power connections to provide for a starting interlock (i.e., pressure cannot be applied from a ground cart until the system electronics is up); (2) an additional solenoid is provided in the solenoid switching assembly to provide redundant shutdown capability as shown in Figure 28.

The hydraulic power supply (Figure 30) consists of a pump with integral cooler, a component module (which houses system components and reservoir) and a solenoid switching valve assembly. The switching valve assembly allows shutdown of the actuator and selection of the third (utility) channel into either actuator sections.

The system is designed for Type II operation per MIL-H-5440G.

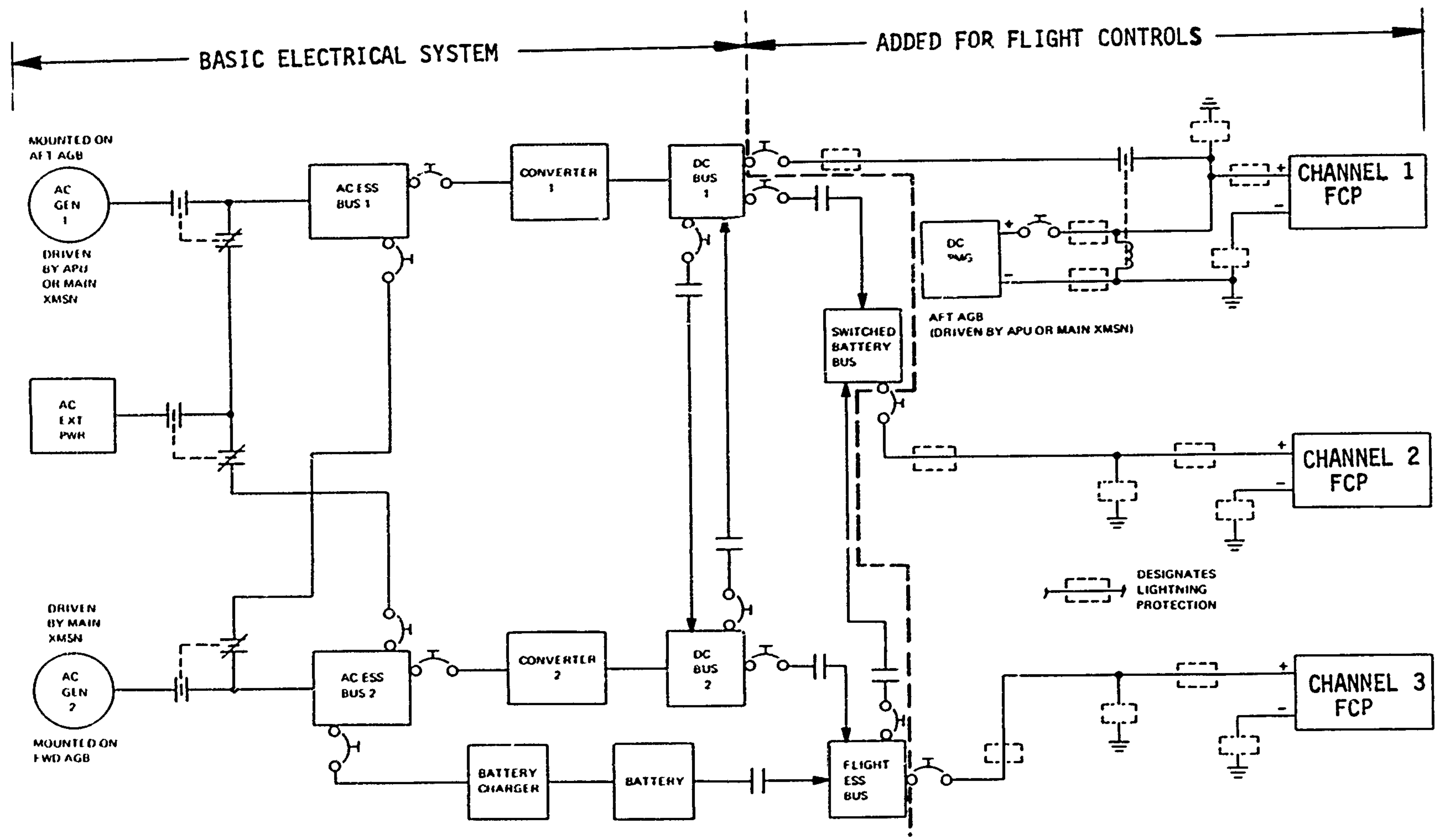


Figure 29. Electrical Power Distribution.

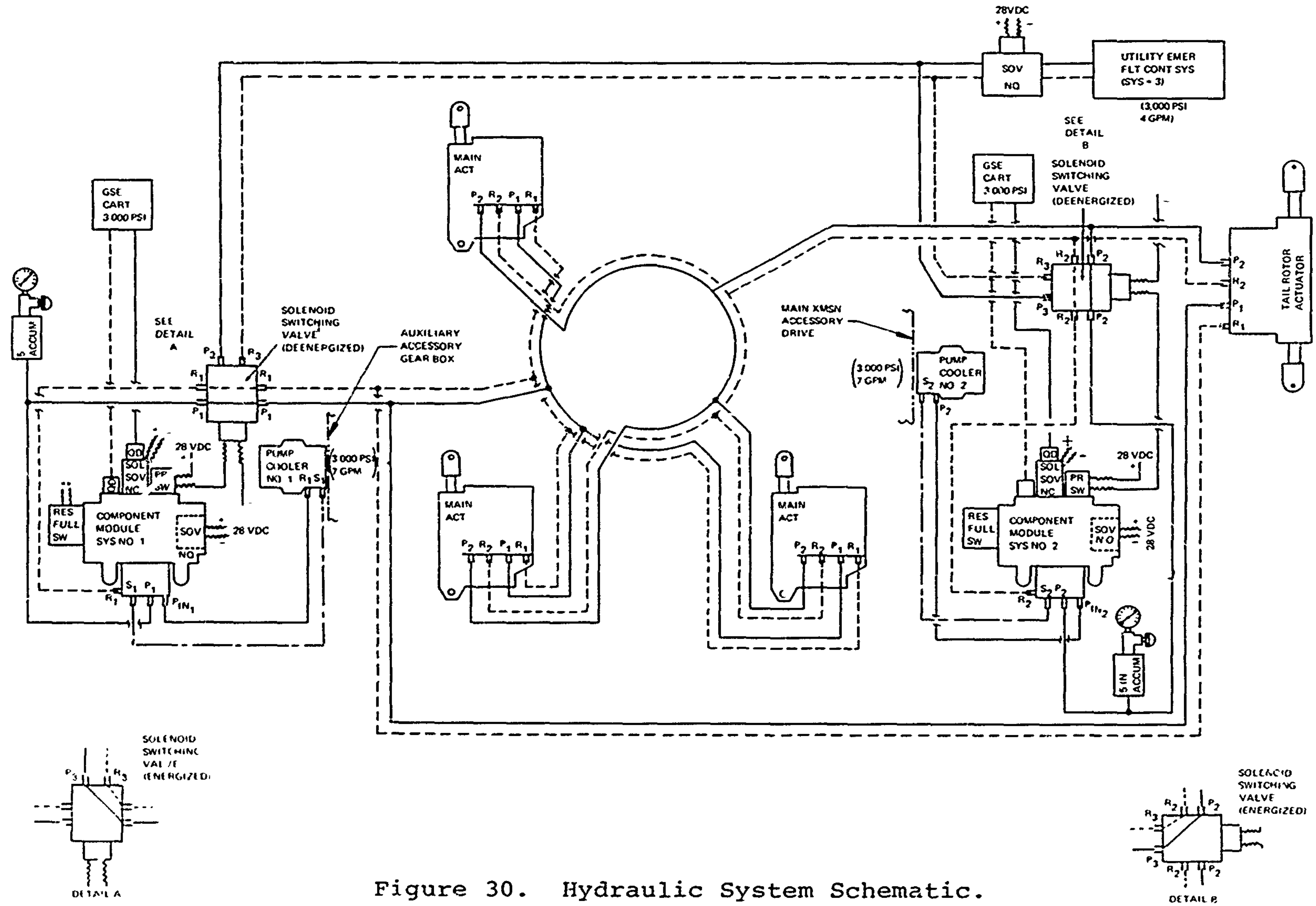


Figure 30. Hydraulic System Schematic.

COMPARATIVE ANALYSES

The following paragraphs cover comparison of the Baseline (FBO) and Alternate (FBW) options with the Dual Mechanical System for the MUT vehicle defined in Reference 1. Areas of comparison include:

1. Handling Qualities
2. Reliability
3. Maintainability/Availability
4. Disability
5. Survivability
6. EMI/Lightning
7. Cost
8. Weight

HANDLING QUALITIES

With automatic systems operating, the handling qualities of the vehicle will be controlled by the AFCS. For new aircraft, this means that all three versions will be essentially similar. Over a period of time the mechanical version will degrade because of looseness which develops within the linkage. Force feel will be degraded as friction increases. For a jam failure of the mechanical version, there will be an attenuation to 50 percent in the control gradient and authority. It will be necessary to apply a breakout force of approximately 60 pounds to free the unfailed side, and an operating force of 3 to 6 pounds to move the control. The non-mechanical versions provide full control from the unfailed side, with no degradation in force feel characteristics.

With AFCS OFF, the mechanical version may provide easier control because of the displacement control available. Boeing simulation studies have shown that the low-displacement (force) type controls are adequate for return to base on loss of AFCS. The probability of complete loss is low because of the fail-operative AFCS configuration provided.

RELIABILITY

Primary Flight Control System (PFCS)

The following is a summary of the reliability of the primary flight control system. These values were calculated for a 1-hour mission.

1. Flight Safety Reliability

Goal:	1 loss/10 million hours
FBO/FBW:	1 loss/28.7 million hours
Dual Mechanical:	1 loss/11.2 million hours

2. Mission Reliability

Goal:	1 ABORT/10,000 hours
FBO/FBW:	1 ABORT/2,880 hours
Dual Mechanical:	1 ABORT/349 hours

3. Maintenance Malfunction Rate

Goal:	1/2,500 hours
FBO/FBW:	1/50 – 1/100 hours
Dual Mechanical:	1/25 – 1/50 hours

Table 6 gives a breakdown of the flight safety failure rate; Table 7 gives a breakdown of the mission abort rate. Figures 31 and 32 are reliability models for the rotor actuators of the baseline and dual mechanical system, respectively. These diagrams show the condition necessary for flight safety loss, i.e., 3 F 3 means three out of three elements failed, or 1 F 2 means one of two failed. The interaction of common elements such as electronics or hydraulic power supply failures is obtained by adding them to the actuator unique failure rate and then subtracting (i.e., F 2 2) the added common failure combination, which is entered once in the failure summation.

It will be noted that the flight safety reliability is dominated by single and dual failures in the rotor control actuators. Mission reliability is dominated by failures in the actuator (in particular in the monitoring portions for the FBO/FBW versions) and by failures in the dualized linkage and actuators for the dual mechanical version.

Figure 33 is a failure model for the actuator failure monitoring function.

Maintenance rates are estimated from previous work on UTTAS FBW/dual mechanical systems. Improvements in FBO/FBW versions result from low-displacement (force) type pilots' controls and use of an electrical linkage in place of a mechanical linkage.

The variation of flight safety reliability goals in the range of 50 to 200 percent would not have an appreciable effect on configuration. At 200 percent there would be a deficiency in the reliability of the mechanical version. A reduction of 50 percent would not remove all of an electrical/optical channel. A triplex system is still required. Reduction of mission and maintenance rates would be appropriate to reflect more achievable goals.

Automatic Flight Control System (AFCS)

For comparison of the three configurations, the system architecture prevents AFCS failures from affecting flight safety. The impact of the AFCS on mission reliability was assessed as follows:

1. For VFR missions, AFCS does not affect mission reliability since those missions may be completed on the PFCS alone.

TABLE 6. PRIMARY FLIGHT CONTROLS FLIGHT SAFETY RELIABILITY COMPARISON

FUNCTION	BASELINE - FBO FAIL/10^6 HOURS	ALTERNATE-FBW FAIL/10^6 HOURS	DUAL MECHANICAL FAIL/10^6 HOURS
Cyclic Controls	1.960×10^{-8}	1.960×10^{-7}	0.00004343
Directional Controls	1.772×10^{-7}	1.772×10^{-7}	0.0010432
Coll Pitch Controls	9.894×10^{-9}	9.894×10^{-9}	0.00004506
Common Electronics and Optics	0.00008738	0.000002136	-
Mechanical Mixing	-	-	0.002826
Main Rotor			
Position 1	0.008631	0.008621	0.02099
Position 2	0.008631	0.008621	0.02099
Position 3	0.008631	0.008621	0.02099
Tail Rotor Controls	0.008631	0.008621	0.02120
Actuator Failure Monitor	0.0002537	0.0002537	-
Electrical Power Supply	5.550×10^{-11}	5.550×10^{-11}	-
Hydraulic Power Supply	0.00001886	0.00001886	0.00001886
TOTAL	.03488	.03476	.08815
MEAN TIME TO FAILURE	28.67×10^6 hr	28.77×10^6 hr	11.35×10^6 hr

TABLE 7. PRIMARY FLIGHT CONTROLS MISSION RELIABILITY COMPARISON

FUNCTION	BASELINE - FBO FAIL/10^6 HOURS	ALTERNATE-FBW FAIL/10^6 HOURS	DUAL MECHANICAL FAIL/10^6 HOURS
Cyclic Controls	0.6485	0.5328	33.353
Directional Controls	0.9113	0.85510	12.504
Coll Pitch Controls	0.3813	0.3238	20.395
Common Electronics and Optics	0.2330	0.04968	-
Mechanical Mixing	-	-	99.552
Main Rotor			
Position 1	61.553	61.511	666.864
Position 2	61.553	61.511	666.858
Position 3	61.553	61.511	666.856
Tail Rotor Controls	61.641	61.599	627.705
Actuator Failure Monitor	30.340	30.340	-
Electrical Power Supply	0.00004408	0.00004408	-
Hydraulic Power Supply	68.403	68.403	68.403
TOTAL	347.2171 Fail/10^6 hr	346.6[illegible]64 Fail/10^6 hr	2863.49 Fail/10^6 hr
MEAN TIME BETWEEN ABORTS	2880 hr	2885 hr	349 hr

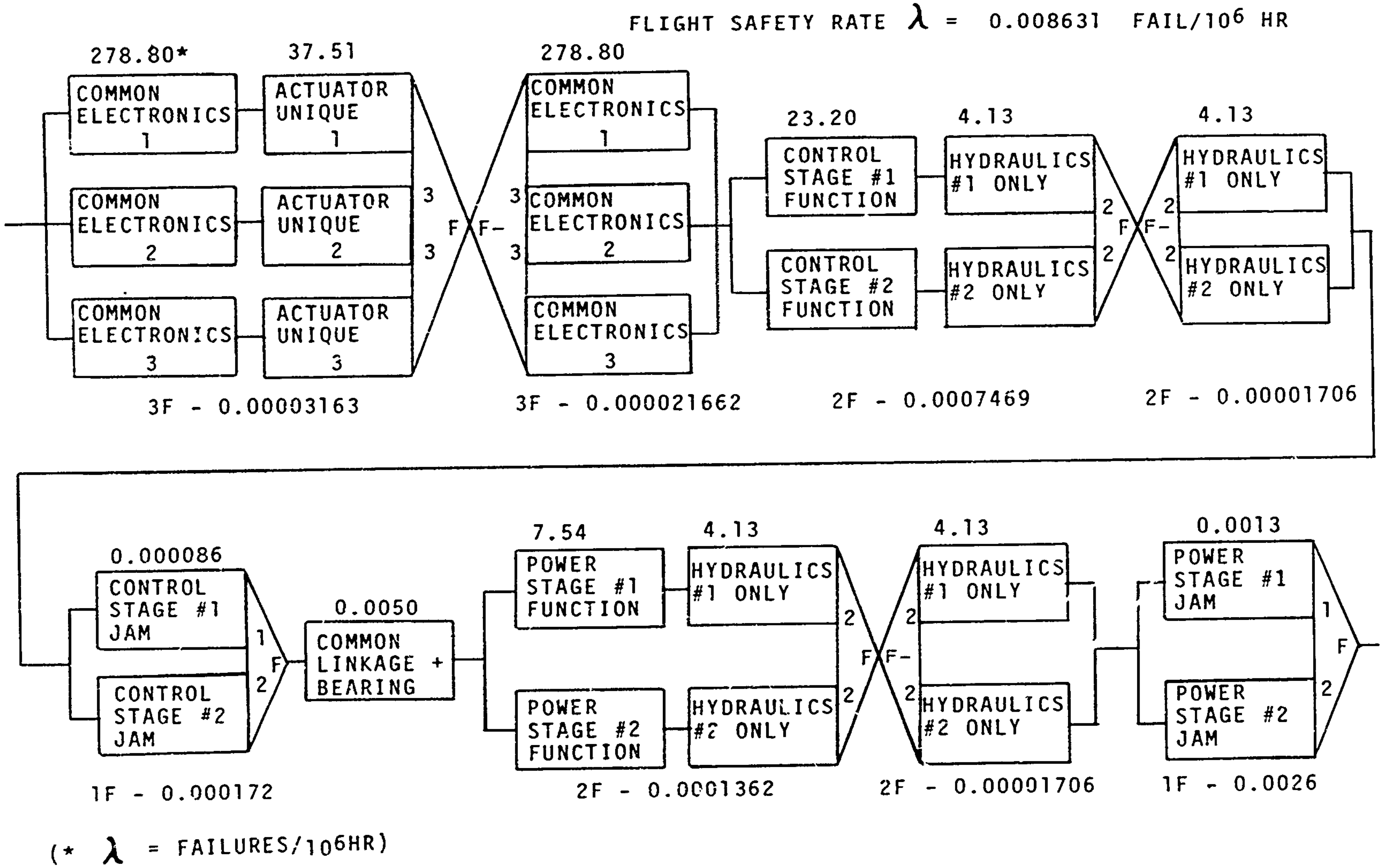
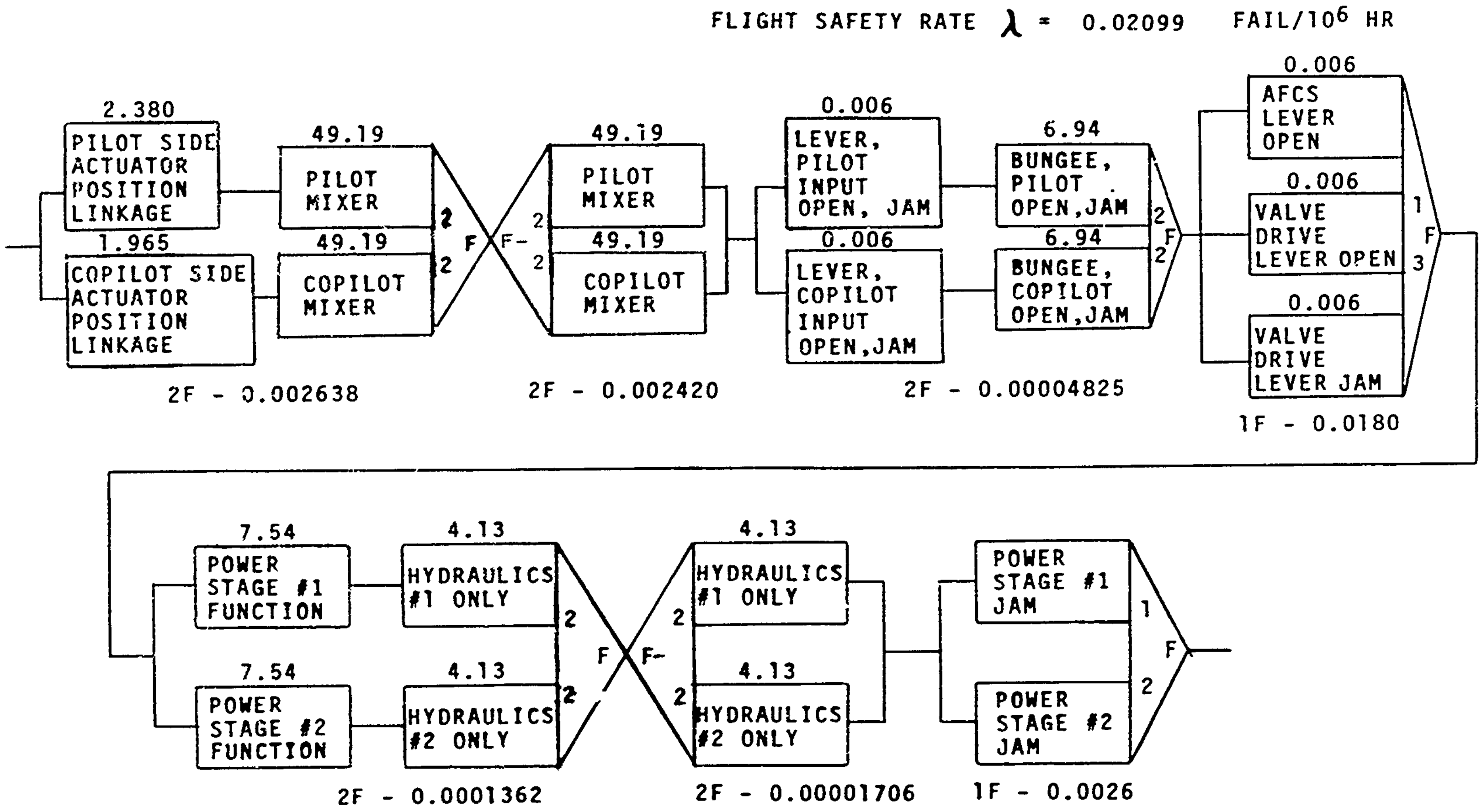


Figure 31. Baseline ASH FCS - Main Rotor Position Flight Safety Reliability Model (One-Hour Mission).



(* λ = FAILURES/10^6 HR)

Figure 32. Dual Mechanical Flight Control System - Main Rotor Actuator Position Flight Safety Reliability Model (One-Hour Mission).

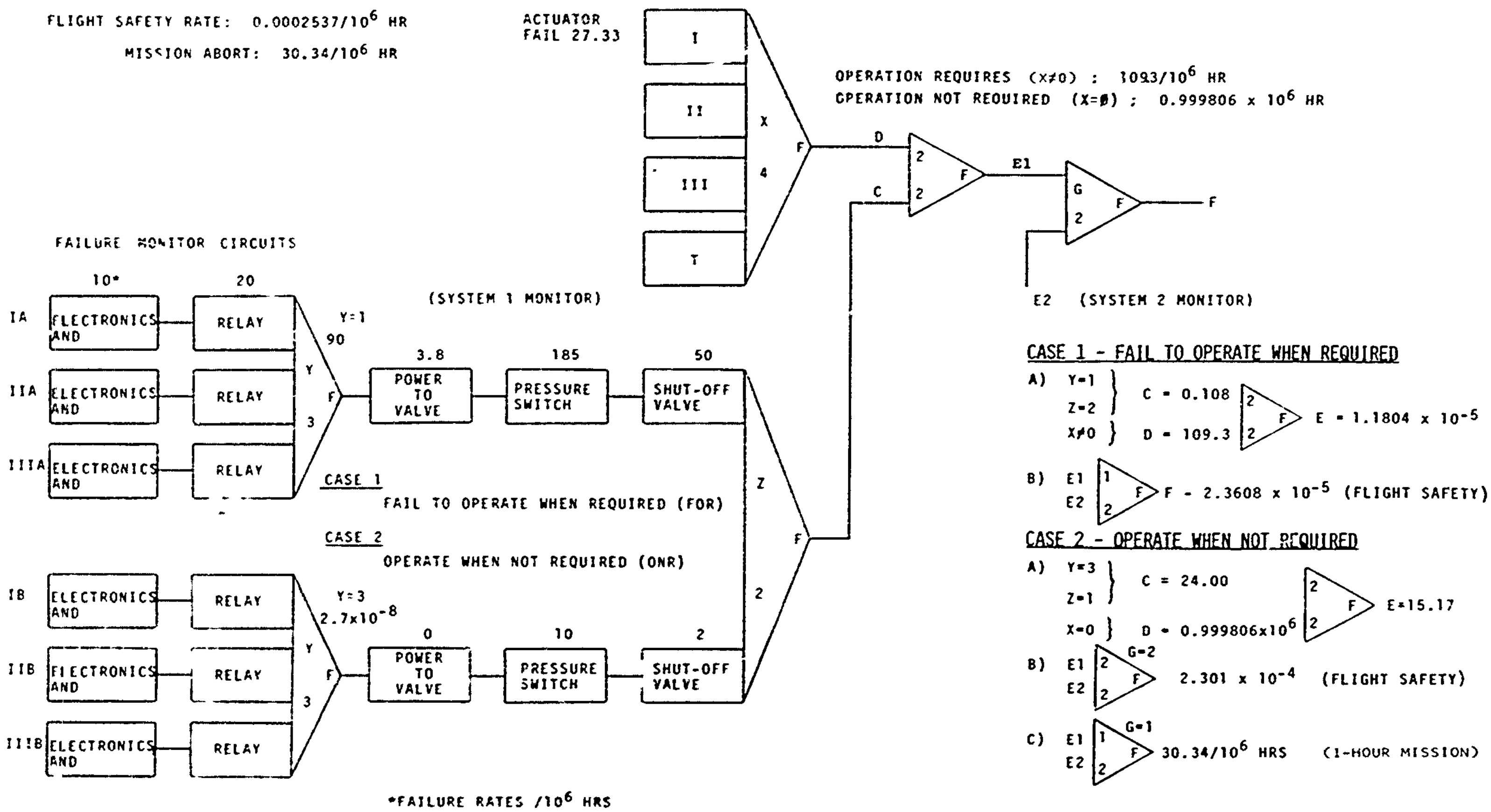


Figure 33. Actuator Monitor Reliability Block Diagram (One-Hour Mission).

2. For missions at night or in IMC, it is assumed that the mission must be aborted when stabilization in either pitch and roll axes, or in yaw axis, is lost. The abort rate for this criteria is 1/3000 hours, which is determined by the first failure of the AFCS computer. The rate for sensor loss is not the primary factor for this case.

3. For missions requiring precision hover at night, it is assumed that the mission must be aborted on loss of any required sensor or loss of one PFCS or AFCS computing channel. The abort rate for this criterion is 1/200 hours.

MAINTAINABILITY/AVAILABILITY

All options will meet the ASH ROC requirements.

The advanced systems will be designed to eliminate any manual preflight check (as was used on HLH designs). Preflight checks will be run automatically on power-up and will not cause actuator motion. Fault isolation will be via the sensor multiplex/test interface unit, which will define the LRU to be replaced. Controllers and actuators are pre-rigged and may be changed without adjustments. After replacement of a failed LRU, the built-in test is run to check the system.

Damaged fiber-optic cables may be replaced on the aircraft using a universal repair "patch cord". This cord would consist of a length (or perhaps two or three lengths) of optical cable, adequate to replace any cable in the system. The three-conductor optical cables would be supplied with pins assembled on each end. The damaged length would be repaired by extracting three pins at each end and replacing these with the patch cord.

In the FBW case, the connection would be made with multiconductor double-shielded cable terminated with connectors. These cables would be stocked as noninterchangeable assemblies in lengths to fit the aircraft. The cost of this approach would be higher than for the FBO approach.

DURABILITY

All system components would be designed for on-condition operation. Rotor actuators will have similar power stage pistons and attaching bearings. There will be little difference in durability for these actuators. The actuators will be designed for an indefinite life with replacement of seals and bearings and repair of power cylinders.

In the mechanical version, bearings must be replaced as they wear out. The FBO/FBW versions do not have bearings that require replacement.

SURVIVABILITY

Table 8 gives a failure assessment of various control system options for the FBO/FBW and dual mechanical versions. The single point vulnerable area of the system consists of the power stage control valve and its immediate drive linkage. This amounts to less than 0 025 ft^2. This area has a negligible effect on vehicle vulnerability.

TABLE 8. ASH FLIGHT CONTROL SYSTEM FAILURE ASSESSMENT

SYSTEM/ELEMENT	1ST FAILURE	2ND FAILURE	3RD FAILURE
ELECTRO-OPTICAL			
Cockpit Inputs	Mission Abort	Attrition	
Signal - Fiber Optics	Operational	Mission Abort	Attrition
Processing - Electronic	Operational	Mission Abort	Attrition
Processing - Mechanical	Mission Abort	Attrition	
Actuators/Lines	Mission Abort	Attrition	
Hydraulic Boost	Operational	Mission Abort	Attrition
FLY-BY-WIRE			
Cockpit Inputs	Mission Abort	Attrition	
Signal - Electrical	Operational	Mission Abort	Attrition
Processing - Electrical	Operational	Mission Abort	Attrition
Processing - Mechanical	Mission Abort	Attrition	
Actuators/Lines	Mission Abort	Attrition	
Hydraulic Boost	Operational	Mission Abort	Attrition
DUAL MECHANICAL			
Cockpit Inputs	Mission Abort	Attrition	
Signal - Mechanical	Mission Abort	Attrition	
Processing - Mechanical	Mission Abort	Attrition	
Actuators/Lines	Mission Abort	Attrition	
Hydraulic Boost	Operational	Mission Abort	Attrition
SINGLE MECHANICAL			
Cockpit Inputs	Mission Abort	Attrition	
Signal - Mechanical	Attrition		
Processing - Mechanical	Attrition		
Actuators/Lines	Mission Abort	Attrition	
Hydraulic Boost	Operational	Mission Abort	Attrition

Figure 34 shows the relative vulnerability of the various systems in terms of probability of kill versus number of hits in a given mission (i.e., between repair actions). FBO/FBW show an advantage because of a smaller presented area and additional redundancy in the linkage path.

EMI/LIGHTNING CONTROL

Control of electromagnetic interference will be achieved by use of fiber optics in the baseline scheme. Other methods that will be used on the remaining electrical lines and in the alternative scheme are:

1. Minimize energy transfer into the LRU by use of: (a) limiting resistors on the input and output of integrated circuits, (b) twisted/shielded cabling for signal leads, (c) filter-pin connections, (d) power input line filters.

2. Provide control for power interruption by use of: (a) long time delay logic discretes, (b) scratch pad memory capacitors, (c) nonvolatile memory.

3. Maintain LRU shielding integrity by: (a) minimizing openings in the LRU, (b) providing a low impedance bond path to structure, (c) locating LRU in a favorable location within the airframe.

The above methods have been used by Honeywell, Incorporated. In tests of digital flight control equipment up to 1,000 amperes/meter no failures occurred. At 1,250 amperes/meter, a change of state of some input discretes occurred; there was no failure.

The flight control processors shown in the report contain protection for the EMI/EMP/lightning strike levels specified in RFP DAA51-79-Q-0129, Control Media Mechanization (Reference 7).

For the fly-by-optics scheme, fibers, emitters, and detectors must also consider the effects of radiation. Available equipment can be selected to meet the threats specified in Reference 7.

COST

Production costs and operation and maintenance costs in early 1980 dollars were estimated for the fleet size as defined below:

1. Fleet Size: 1,450 helicopters

2. Production Rate:
   (a) 2 helicopters per month for 6 months.
   (b) 8 helicopters per month for 6 months.
   (c) 15 helicopters per month for 4 months.
   (d) 20 helicopters per month for remainder.

3. Aircraft Life: 15 years

4. Flying Hour Program: 480 hours per year.

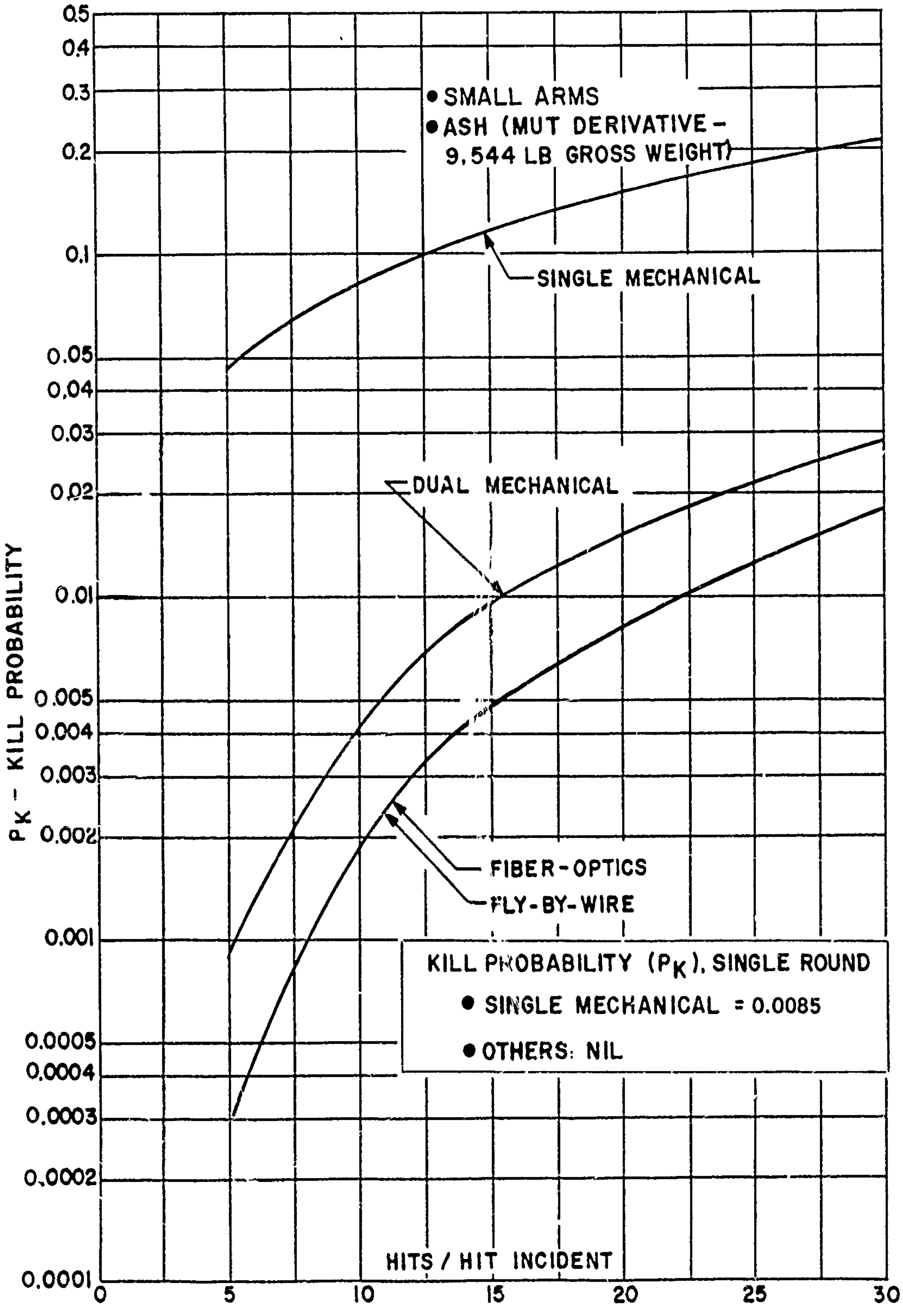


Figure 34. Flight Control System Vulnerability Comparison.

Costs are shown in Tables 9 and 10. Considering the estimation methods, the total are essentially the same, showing that the advanced systems are competitive in cost with the dual mechanical approach. For a new aircraft design, development cost would also be similar for the three schemes.

Cost details are given in Volume II.

WEIGHT

Table 11 gives the estimated weight for the three competing systems. In the view of Boeing Vertol Weights Engineering, these weights are not fully optimized relative to the UTTAS design. The weights for the main rotor actuator reflect the current status and goal suggested by Boeing Vertol Weights Engineering.

The table shows an advantage of 134-157 lb for FBO and 106-128 lb for FBW over dual mechanical. By removing the common elements, i.e., the horizontal stabilizer actuator, hydraulic supplies, upper rotating controls, sensor multiplex unit, and pitot static system, which total 248 lb, the FBO weight is 71.2 percent of dual mechanical and FBW weight is 76.6 percent of dual mechanical.

11
B

TABLE 9. FLIGHT CONTROL SYSTEM COST COMPARISON

	FLY-BY-OPTICS	FLY-BY-WIRE	DUAL MECH
Production Cost	$377,010,150	$372,383,200	$370,057,400
Operations & Maint Cost	87,039,577	72,223,313	81,372,009
TOTAL	$464,049,727	$444,606,513	$451,429,409

TABLE 10. FLIGHT CONTROL SYSTEM OPERATIONS & MAINTENANCE COST

	FLY-BY-OPTICS	FLY-BY-WIRE	DUAL MECH
AVUM	$ 5,943,205	$ 5,883,635	$ 11,233,780
AVIM	14,318,720	12,699,770	7,874,370
DEPOT	47,107,376	33,918,513	39,976,169
SCRAPPED	19,670,276	19,721,395	22,287,690
TOTAL O & M COST	$ 87,039,577	$ 72,223,313	$ 81,372,009

TABLE 11. FLIGHT CONTROLS WEIGHT COMPARISON - BASED ON 9,544-LB GW MUT

ITEM	DUAL MECHANICAL	FLY-BY-OPTICS	FLY-BY-WIRE
COCKPIT CONTROLS (INCLUDING TRANSDUCER, SPRINGS, DRIVE ACTUATORS, STRUCTURE)	102	72	72
CONTROL ELECTRONICS	60	80	71
CONTROL RUNS, ELECTRICAL/ OPTICAL	37	32	69
MECHANICAL CONTROL RUNS	120	36	36
MAIN ROTOR ACTUATORS	138 (*115)	108	108
TAIL ROTOR ACTUATOR	28	24	24
HORIZONTAL STAB, ACTUATOR, AND SUPPORTS	22	22	22
ELECTRICAL POWER SUPPLY	-	5	5
HYDRAULIC POWER SUPPLIES	95	95	95
UPPER ROTATING CONTROLS	109	109	109
SENSOR/MULTIPLEX UNIT	16	16	16
AFCS INSTALLATION	13	2	2
3RD PITOT SYSTEM	6	6	6
INSTALLATION BRACKETS	48	30	30
TOTALS	794 LB	637 LB	666 LB
* GOAL	771 LB *		

CONCLUSIONS

Table 12 presents a summary of the comparisons given in the previous section.

Based on these comparisons, Boeing Vertol concludes that the fly-by-optics offers the best approach for a future advanced flight system. Principal areas needing further development are:

1. Optical Transducer and its interfaces.

2. Computer architecture permitting effective validation.

Optical transducers are in development at Boeing Aerospace Company and other locations. Solution of the validation problem is also in work. Boeing's separated and in-line monitored digital primary system will allow software validation with high confidence because it limits the scope of the flight safety programming to the primary control laws. Once these control laws have been validated, changes in automatic functions cannot affect them. This approach offers marked contrast to other systems, which include all functions in a large processor. The fly-by-wire alternative also offers an improvement over the dual mechanical approach. Development risks are lower than with the optical system because the conventional electrical transducers and signalling used are similar to those flown on the HLH/347 demonstration aircraft.

The use of a digital architecture for the flight safety processing is still a risk. If the validation of a digital primary system cannot be achieved with adequate confidence, use of an analog PFCS integrated in the same LRU with a digital AFCS would be a best alternative. In this case, large-scale integration (LSI) techniques would be used to reduce the cost, weight and size of the analog portion.

TABLE 12. ANALYSIS SUMMARY

PARAMETER	DUAL MECHANICAL	COMPARED TO MECHANICAL	
		FBW	FBO
HANDING QUALITIES		X	X
RELIABILITY			
FLIGHT SAFETY	MEETS REQUIREMENT	X	X
MISSION		X	X
MAINTAINABILITY		X	X (BEST)
AVAILABILITY		X	X (BEST)
DURABILITY		X	X
SURVIVABILITY - SMALL ARMS		X	X
EMI/EMP/LIGHTNING TOLERANCE	X (BEST)		X
COSTS			
PRODUCTION	NONE HAS A SIGNIFICANT ADVANTAGE		
LIFE CYCLE			
WEIGHT		X	X (BEST)

X ADVANTAGE

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APPENDIX A

ADVANCED SCOUT HELICOPTER FLIGHT CONTROL SYSTEM SPECIFICATION

OBJECTIVE

The Applied Technology Laboratory has contracted with Boeing Vertol to study a new fly-by-wire flight control system for the Advanced Scout Helicopter (ASH). The purpose of this preliminary development specification is to solicit vendor technical and budgetary cost response for the development of the fly-by-wire flight control system.

The flight control system comprises the primary flight control system (PFCS) and the automatic flight control system (AFCS). The PFCS is considered to include the control transducers, electronics, rotor actuators, and control panels. The AFCS is considered to include the AFCS sensors and the AFCS electronics.

There shall be two concepts. The baseline triplex model shall use fiber-optic force transducers to measure cockpit control and actuator positions. The alternate triplex model shall use LVDTs (linear variable differential transducers) instead of the optical transducers.

SYSTEM REQUIREMENTS

Both the concepts (baseline and alternate) must meet the requirements which will be defined in each of the following categories:

GENERAL FUNCTIONAL REQUIREMENTS

Flight Control System Dynamics

Figure A-1 shows the cockpit controls with force controllers. Each force controller shall output three redundant signals, which go into three redundant channels. Each channel has two dual paths to allow self-monitoring. Figures A-2, A-3, A-4, A-5, and A-6 show the overall system dynamics. Figure A-4 shows details of the flight control processor. Details of the fiber-optic transducer interface are Boeing proprietary information. They are presented in Volume II.

The primary control system incorporates two control paths, which include separated processors. These are interfaced with a third processor that performs the AFCS computations. This AFCS processor receives triplex sensor and discrete inputs from both external sensors and the primary control system. Signals from the two adjacent channels are sent on dedicated optical links. These are voted and failure detected as described in the AFCS Interface section. The AFCS processor also receives simplex signal inputs and discretes via optical links. The calculated outputs of the AFCS processor are supplied to each primary control processor where they are voted and failure detected. Desired response to pilot and AFCS signals is calculated, limited, and supplied as an actuator command to be compared with actuator feedback (see Figure A-5).

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- Primary system

Figures A-7, A-8, A-9, and A-10 present the primary system transfer functions They also show the mixing between pilot and copilot signals and the shutoff logic in the case of hardover. Figure A-11 shows the cyclic control mixing and cumulative limits.

- AFCS

1. Figures A-12, A-13, and A-14 show the basic triplex AFCS: longitudinal, lateral, and directional.

2. Figures A-15, A-16, and A-17 show the dynamics of the attitude hold and the hover hold modes.

3. Figures A-18 and A-19 show the cyclic decoupler, and the horizontal stabilizer trim of the AFCS.

AFCS Interface

The following shall constitute the interfaces of the AFCS:

- Inputs

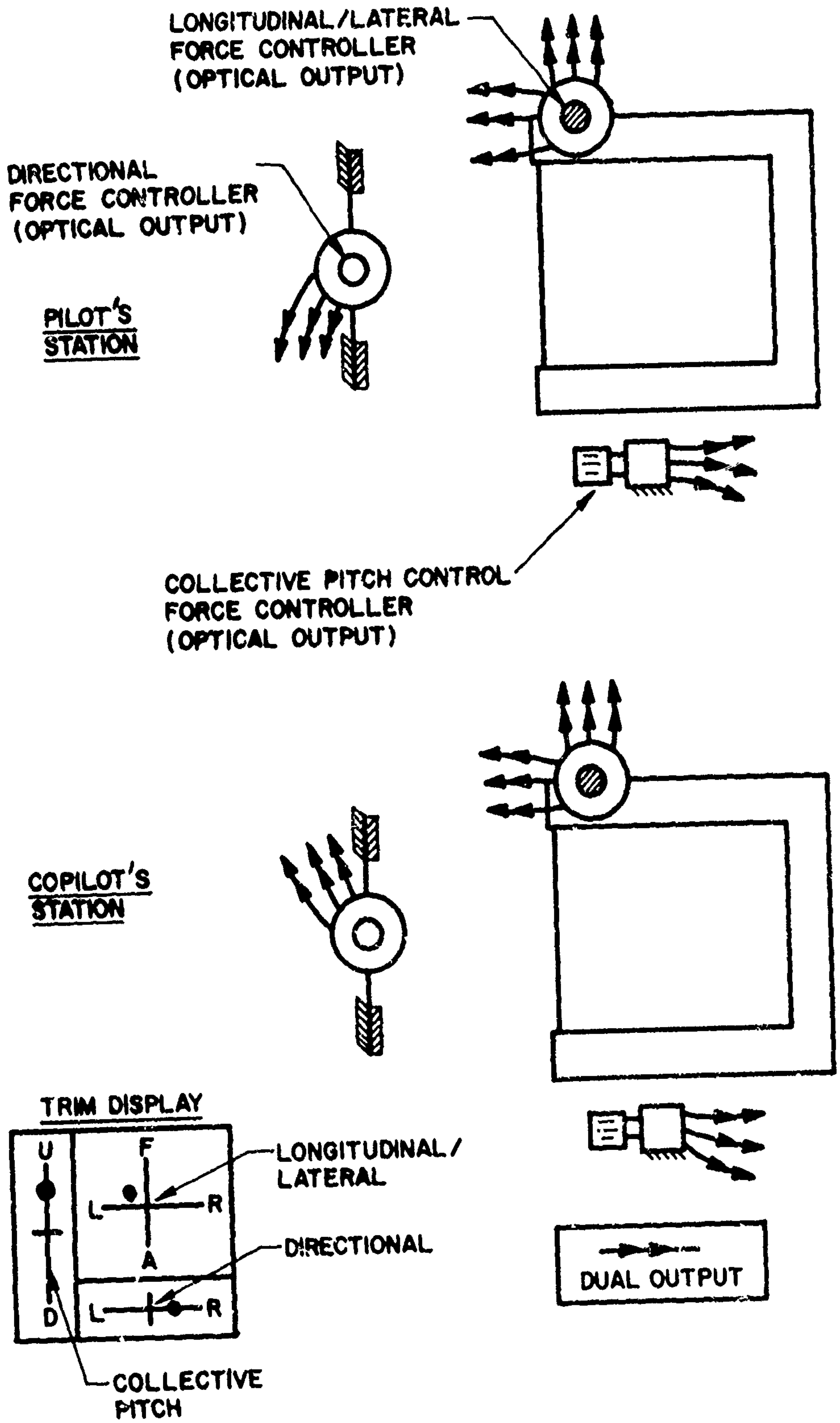

Figure A-1. Baseline Cockpit Control Concept.

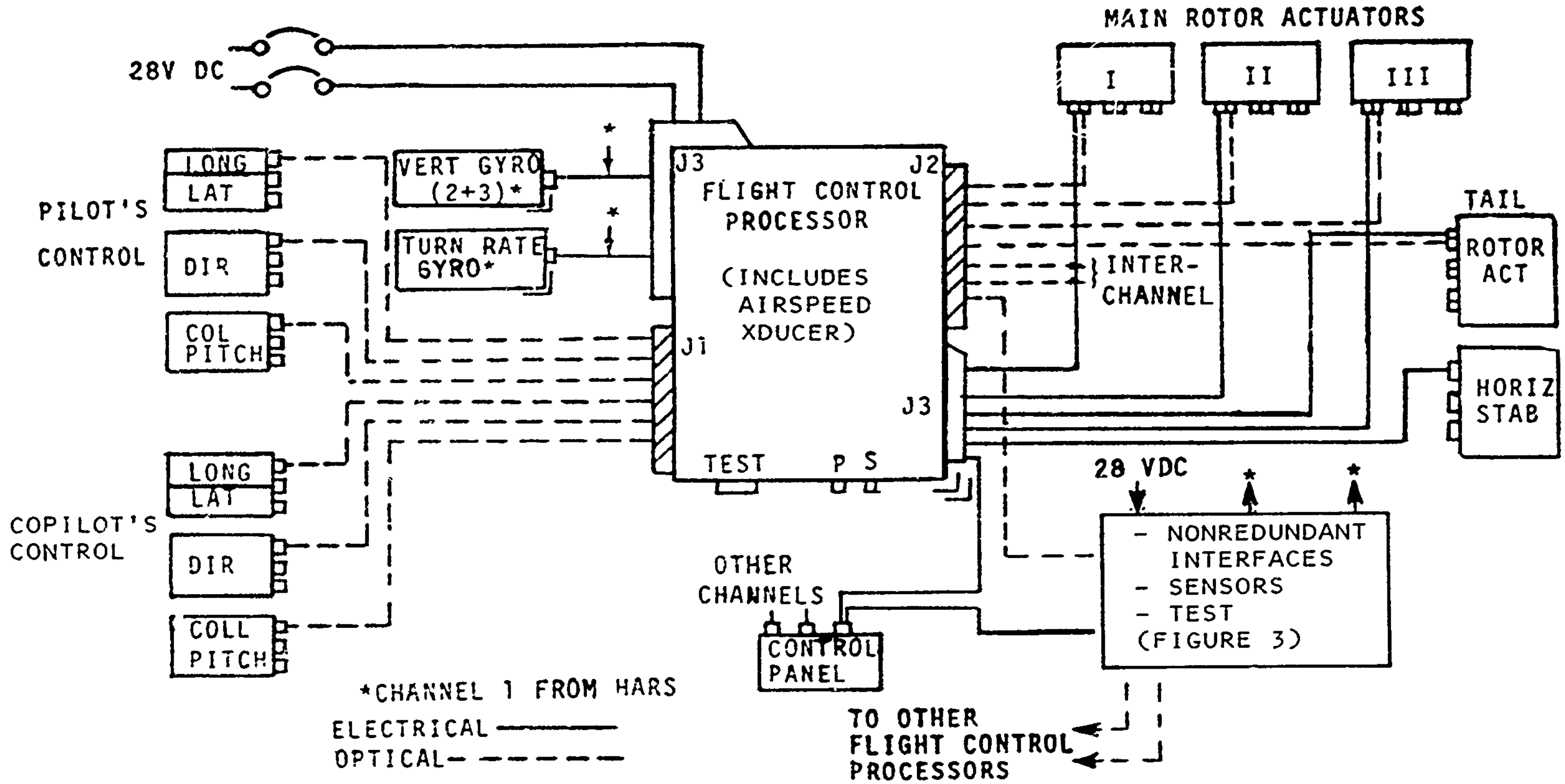

Figure A-2. Baseline ASH Flight Control System - Equipment Diagram.

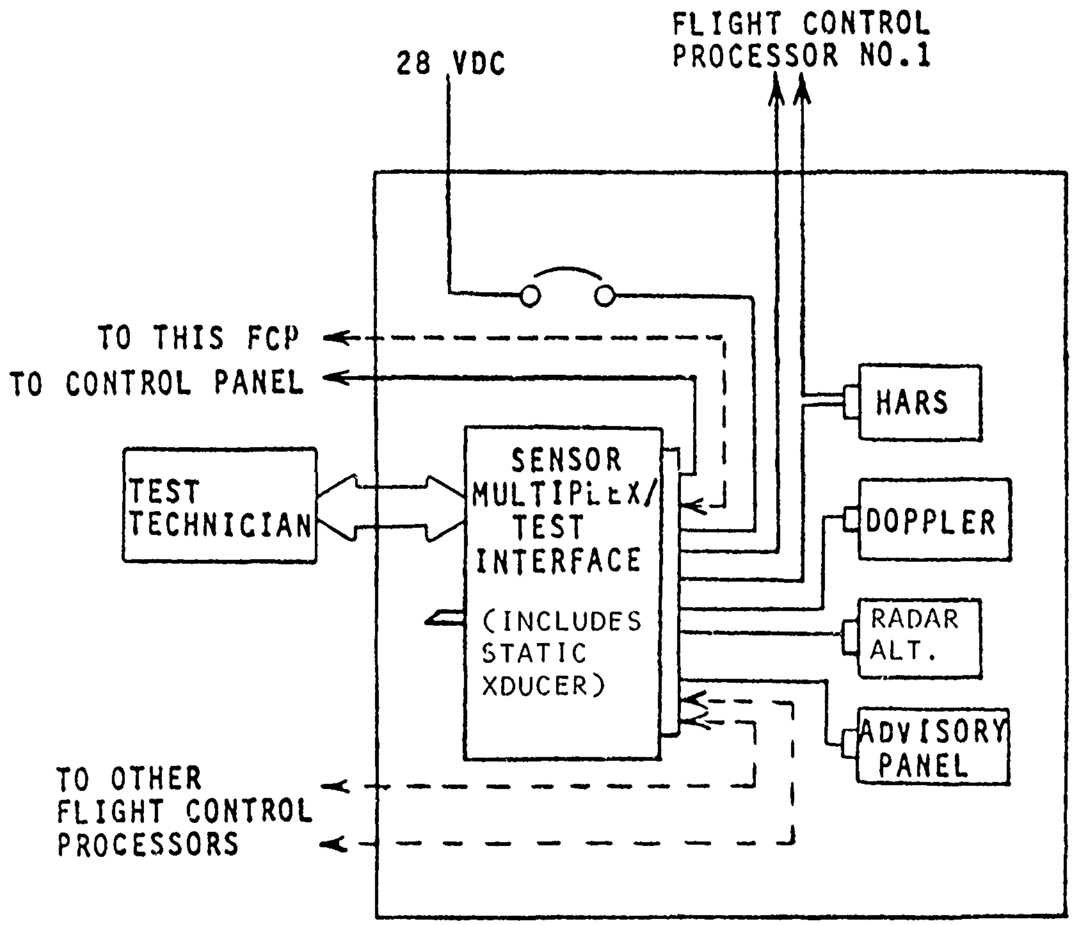

Figure A-3. Nonredundant Interfaces - Sensors/Test.

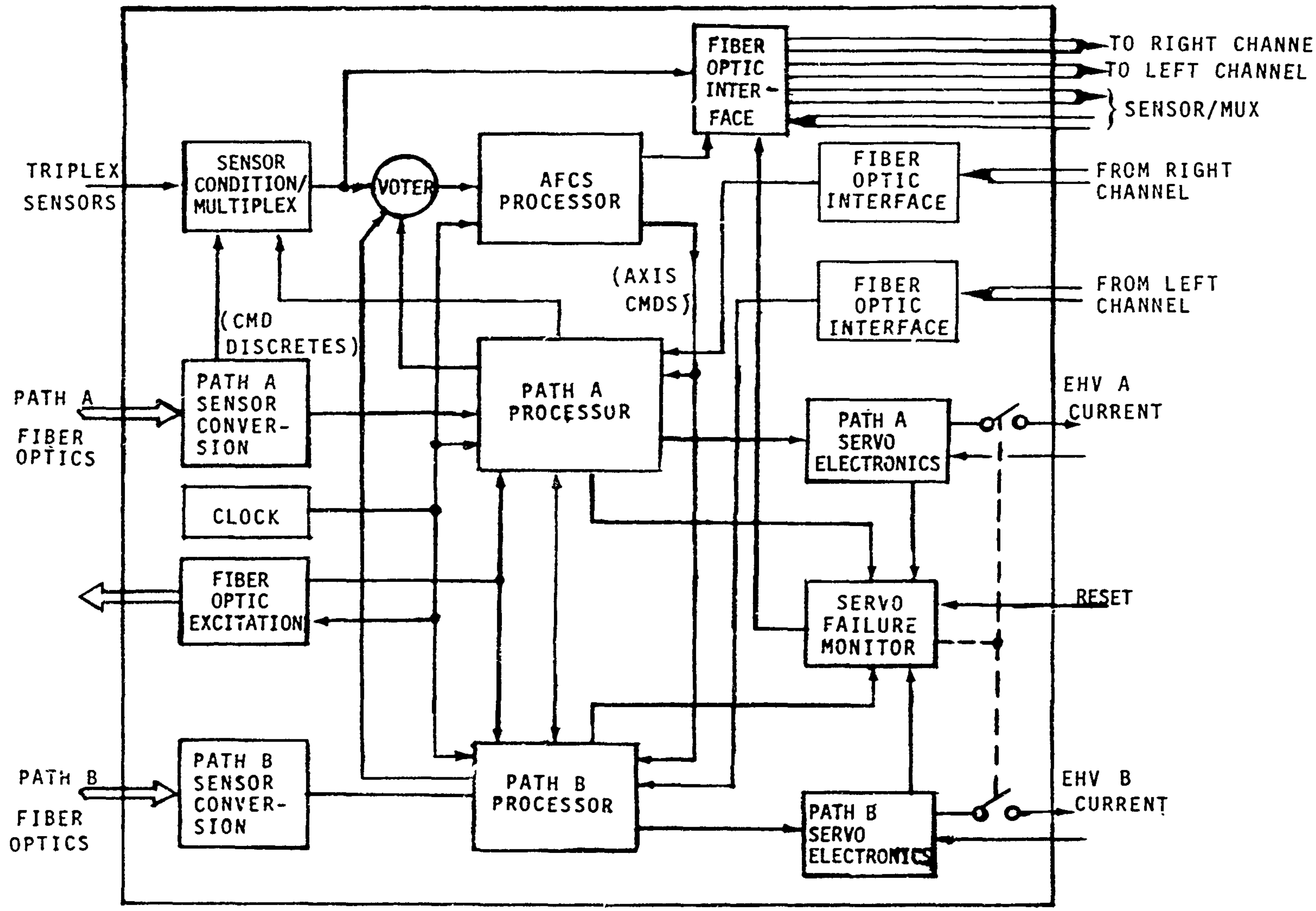

Figure A-4. Baseline Flight Control Processor - Block Diagram.

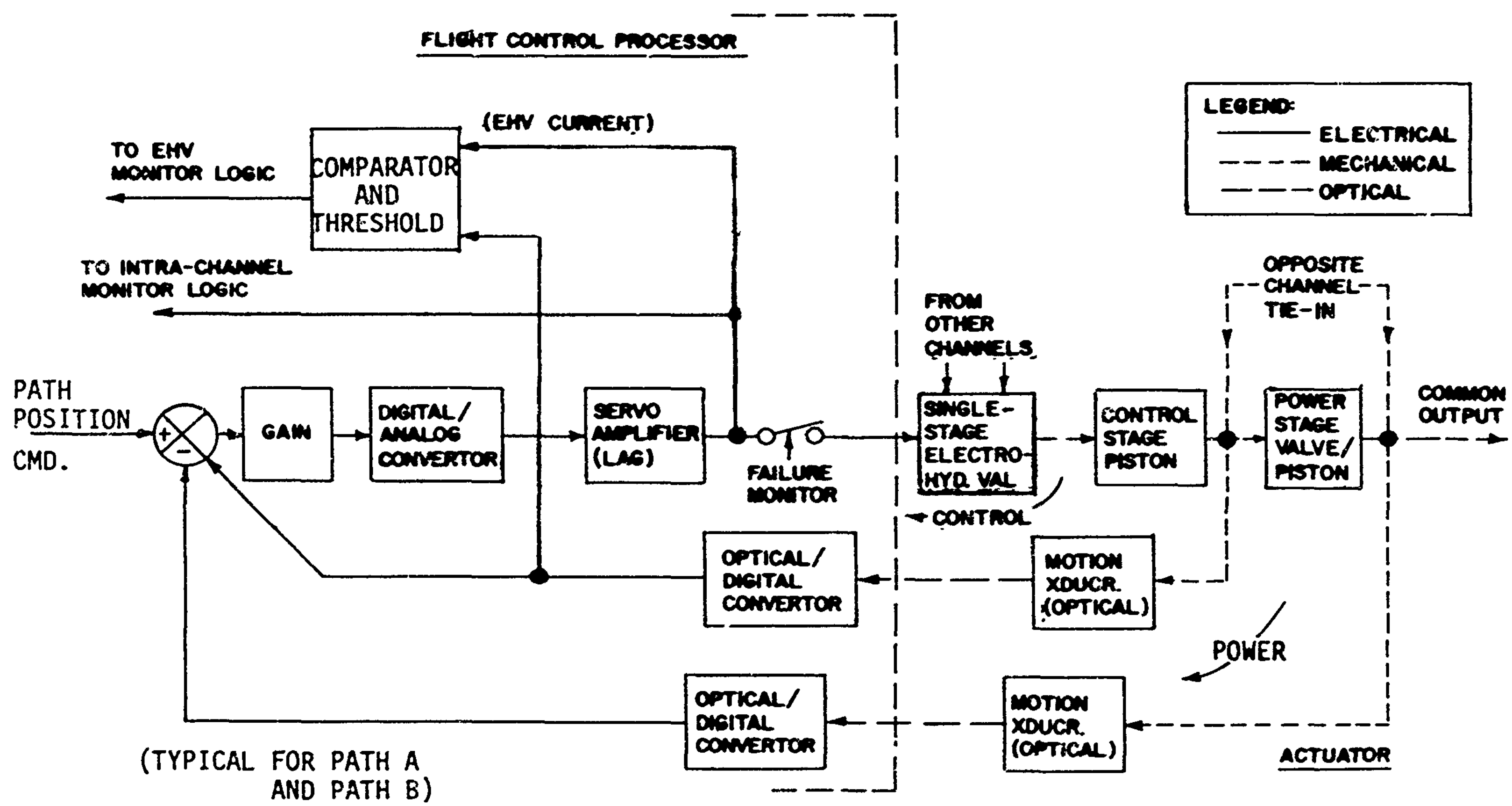

Figure A-5. Baseline Rotor Control Actuator-Servo Loop/ Failure Monitor.

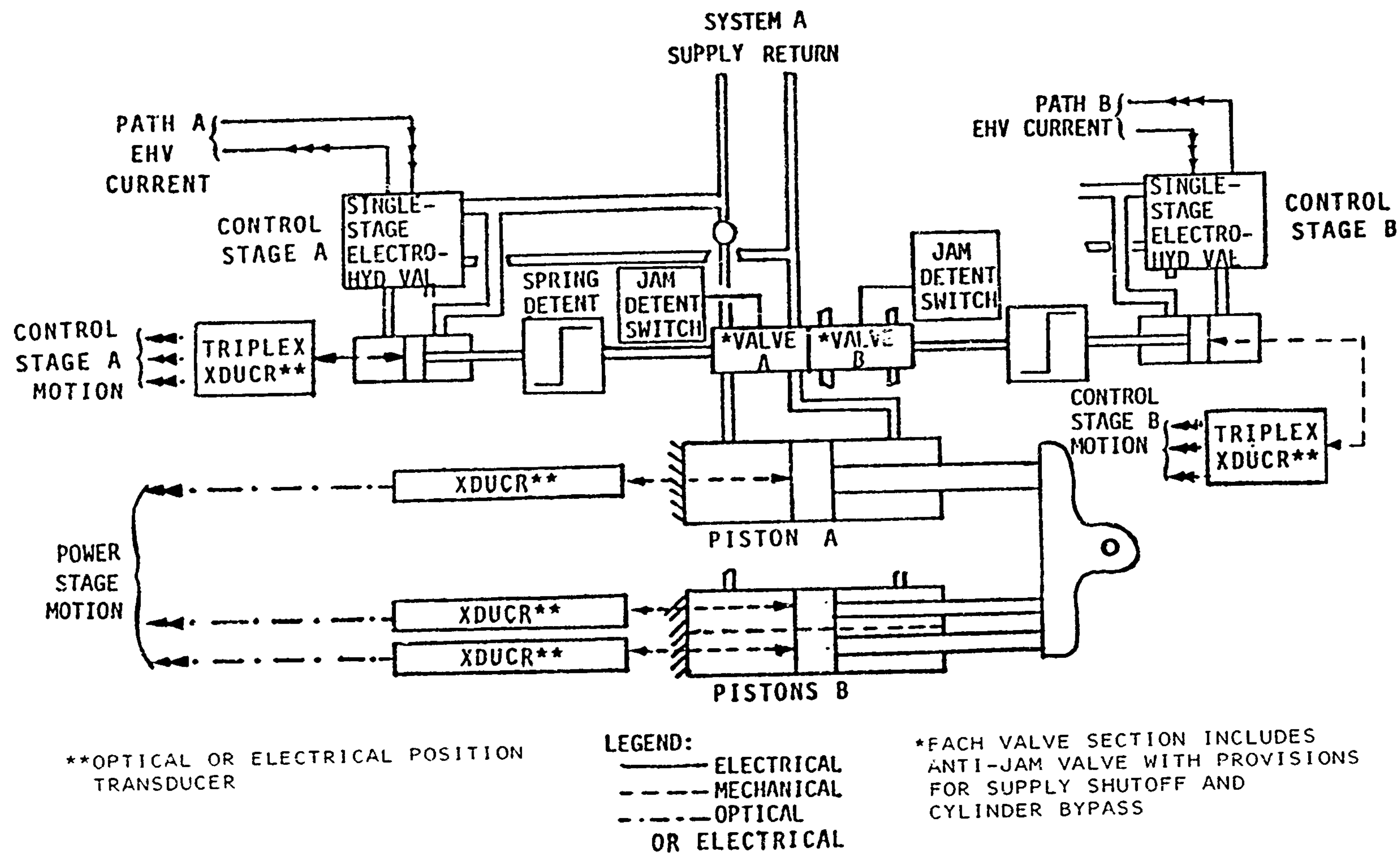

Figure A-6. Rotor Control Actuator.

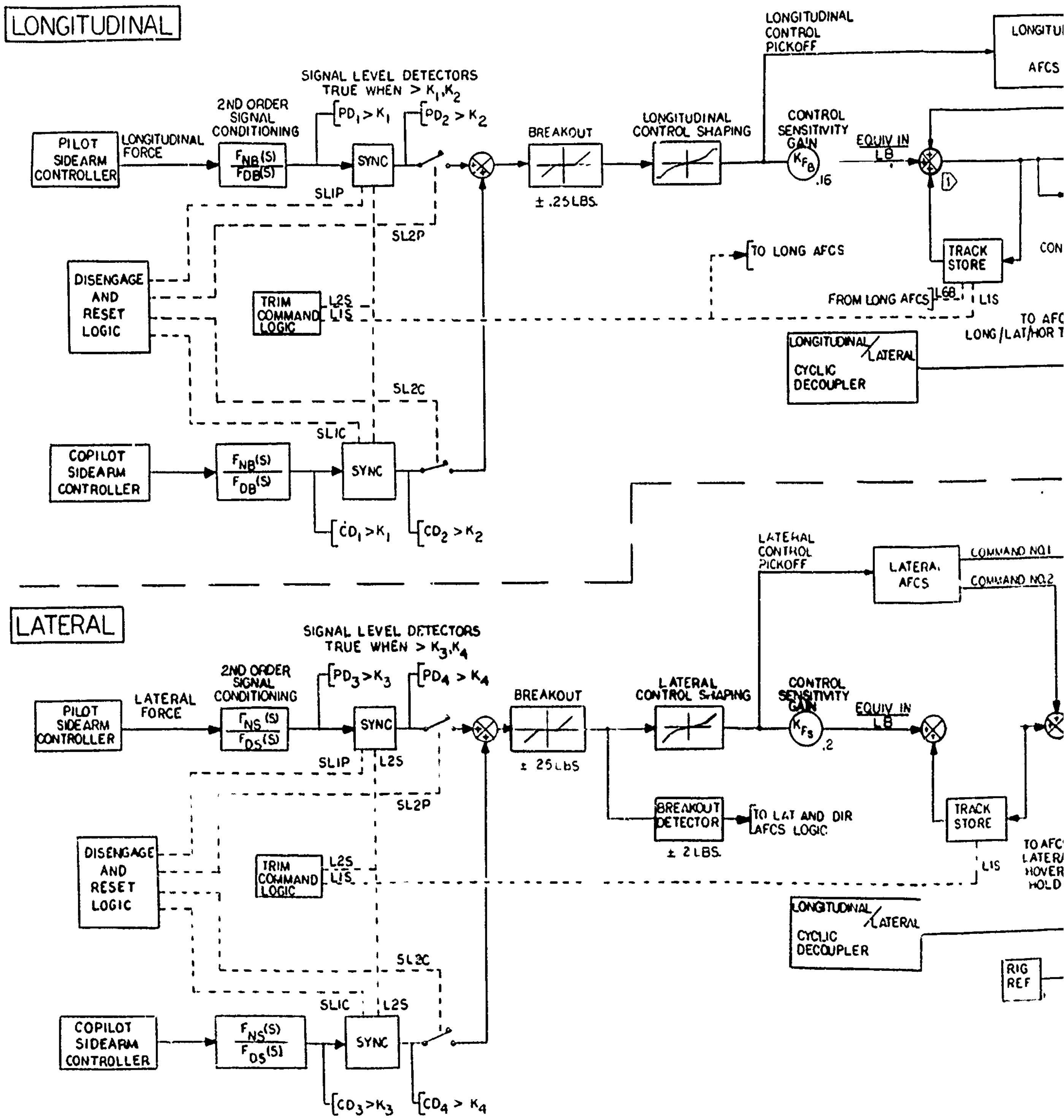

Figure A-7. PFCS Block Diagram - Longitudinal/Lateral Axes.

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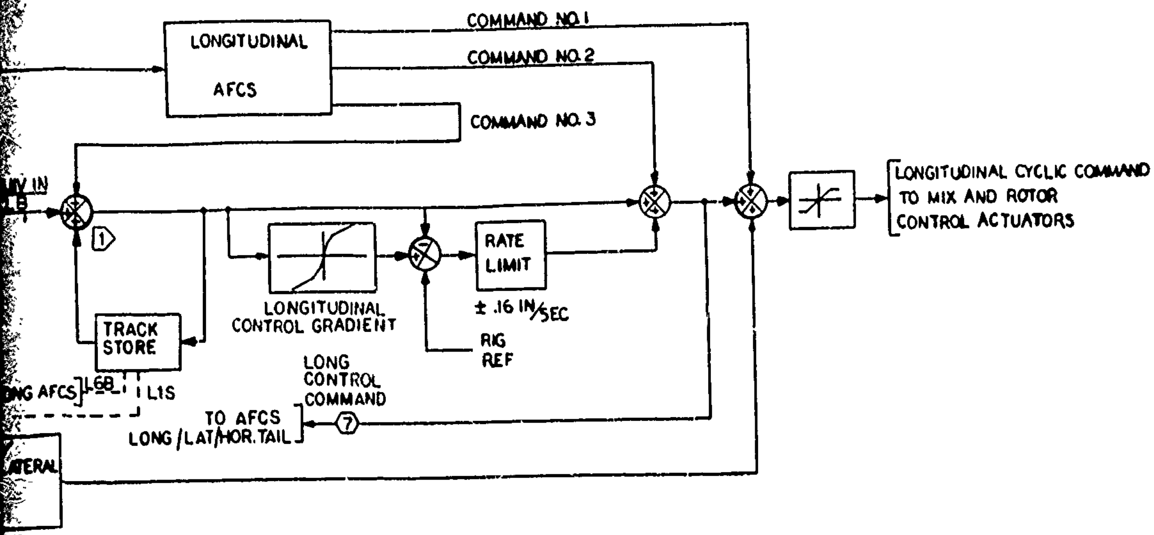
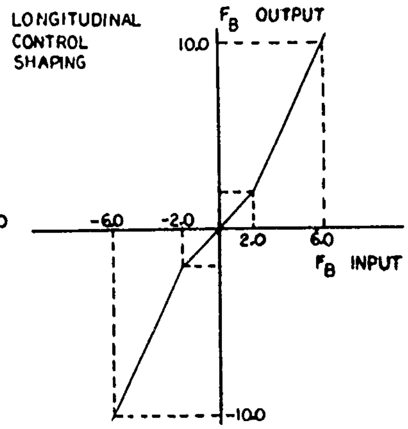
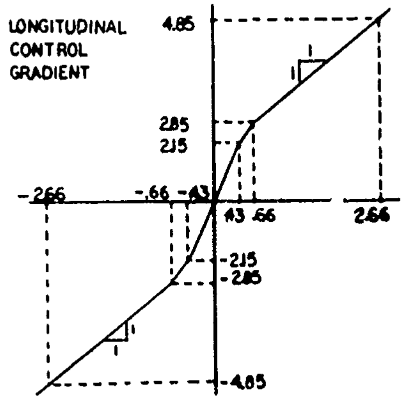
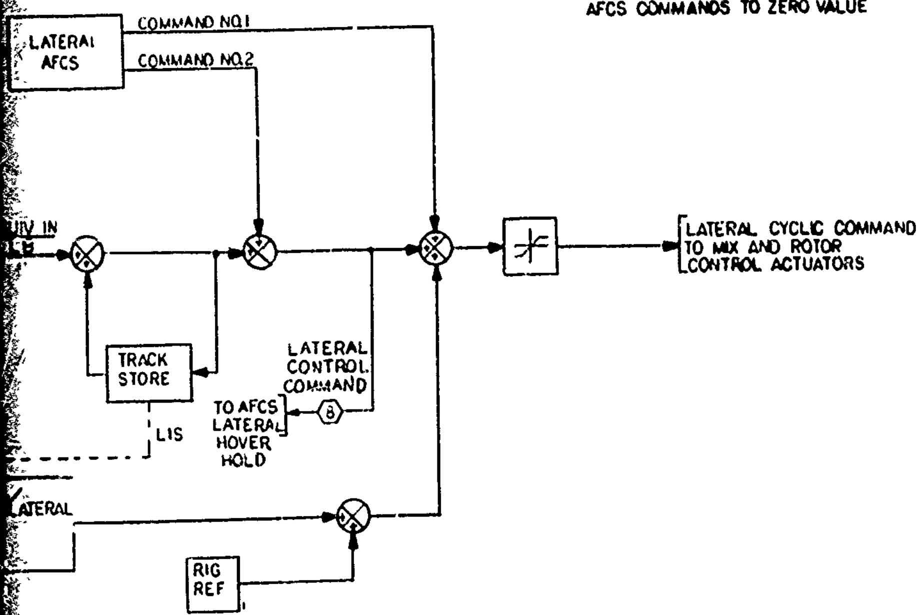
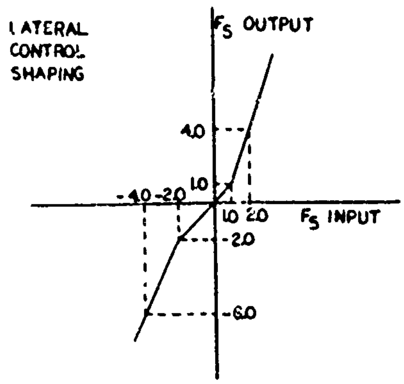






NOTE:
1. TRACK STORE FUNCTION IS VECTORED IN COMPUTATION CYCLE BEFORE SUMMATION TO PERMIT TRANSIENT-FREE TRIMMING OF CONTROLLER FORCES AND SYNCHRONIZATION OF AFCS COMMANDS TO ZERO VALUE





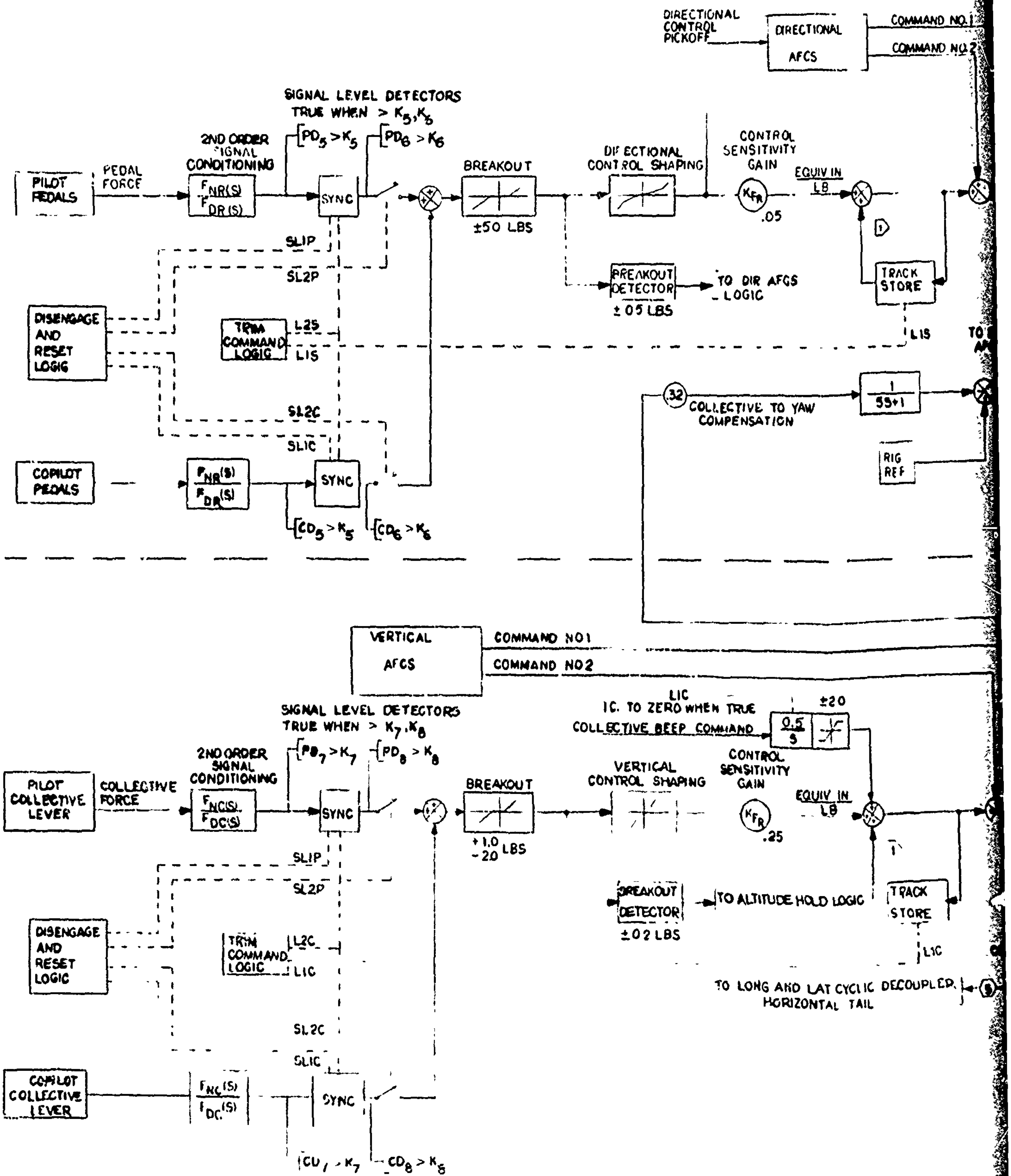


Figure A-8. PFCS Block Diagram - Directional/Vertical Axes.

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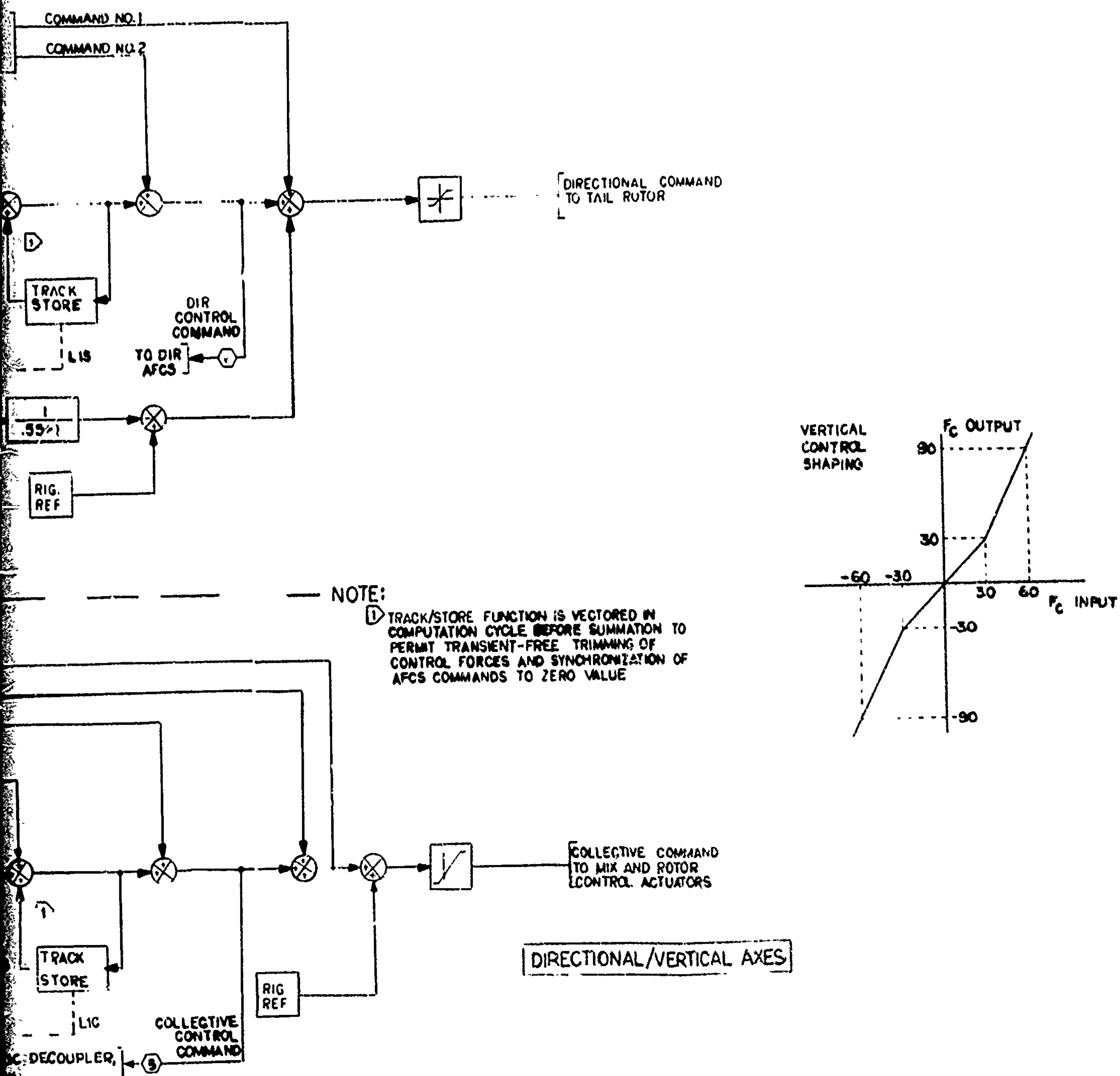


DIRECTIONAL/VERTICAL AXES

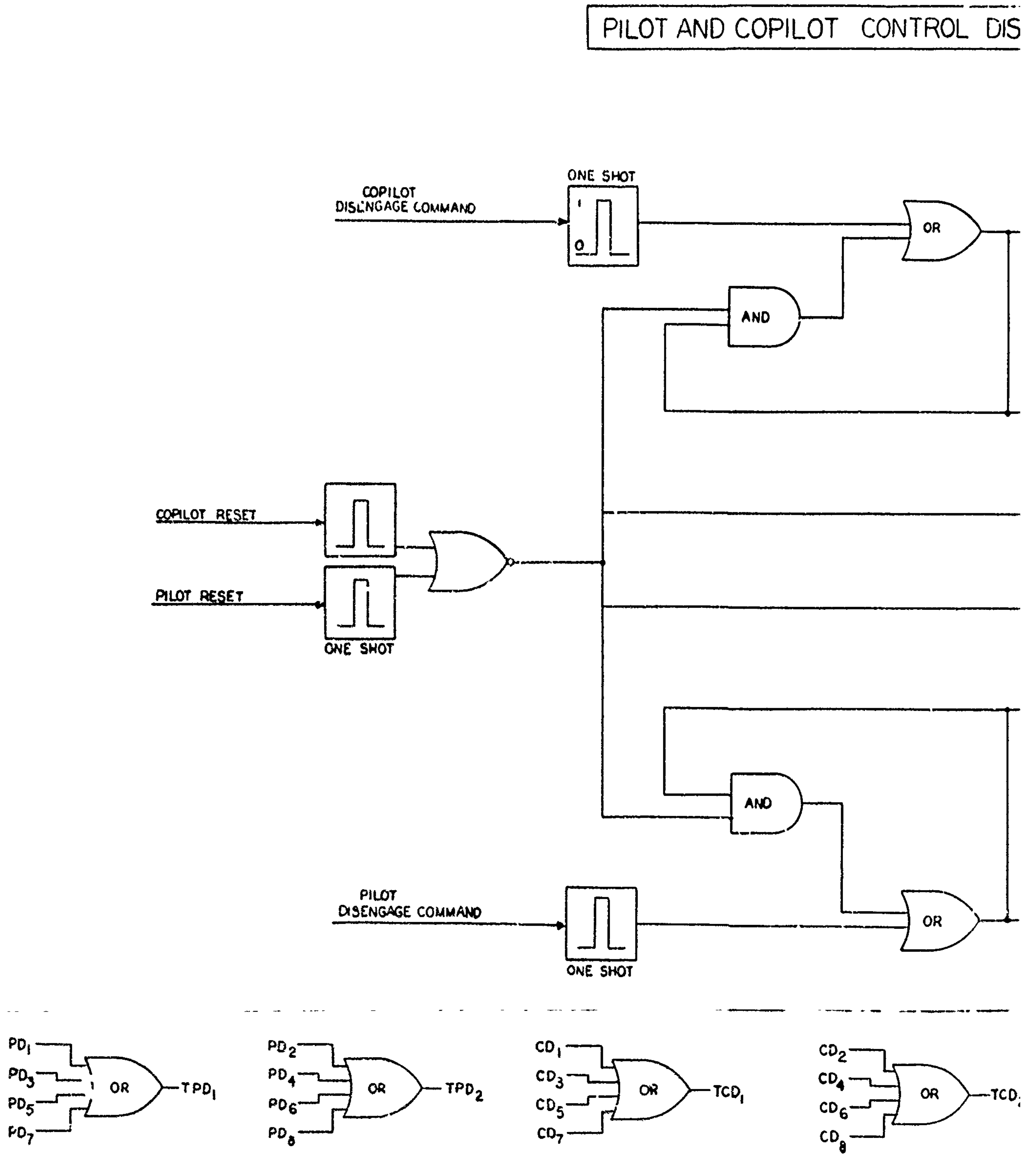


Figure A-9. Control Disengage and Reset Logic.

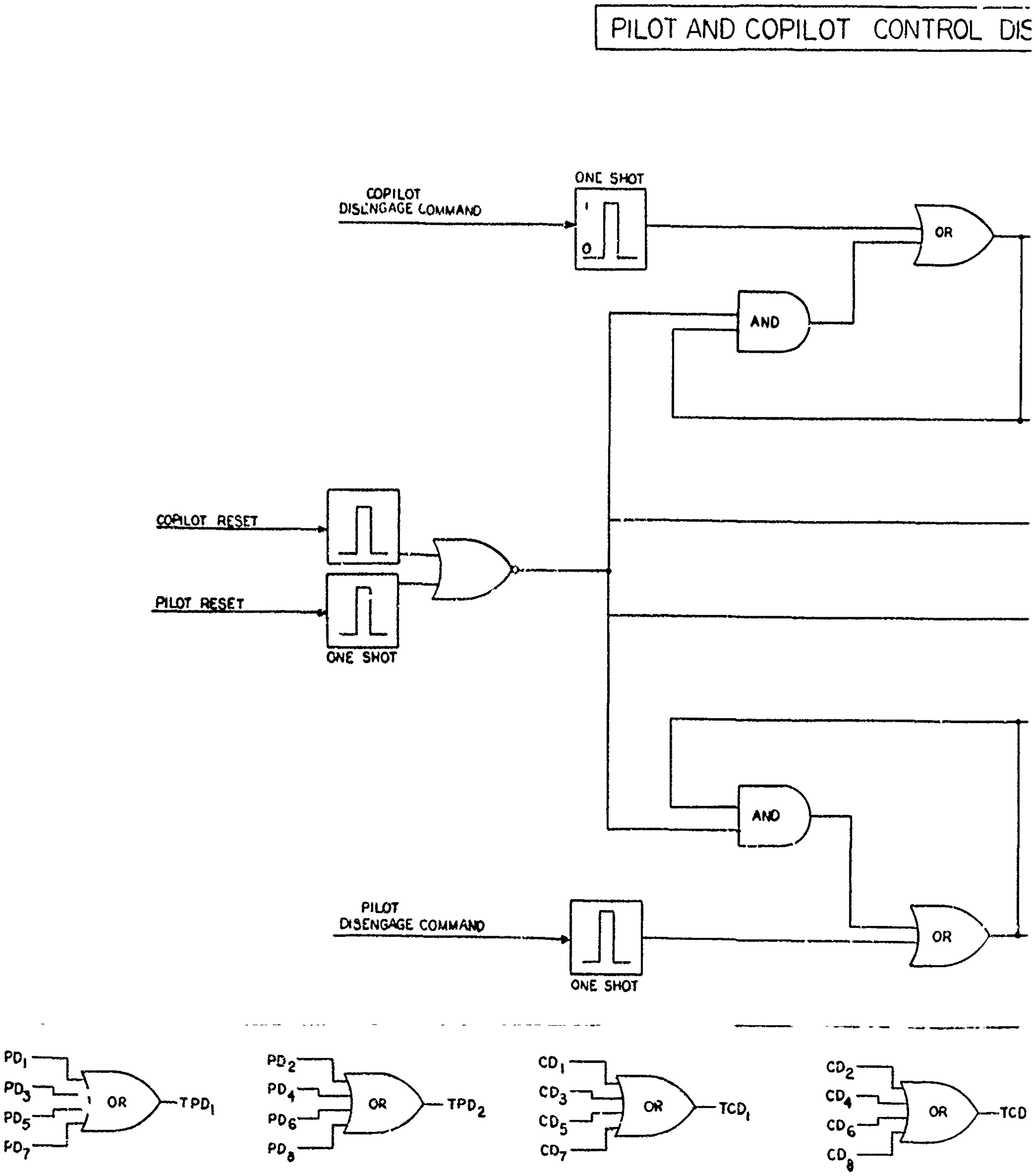
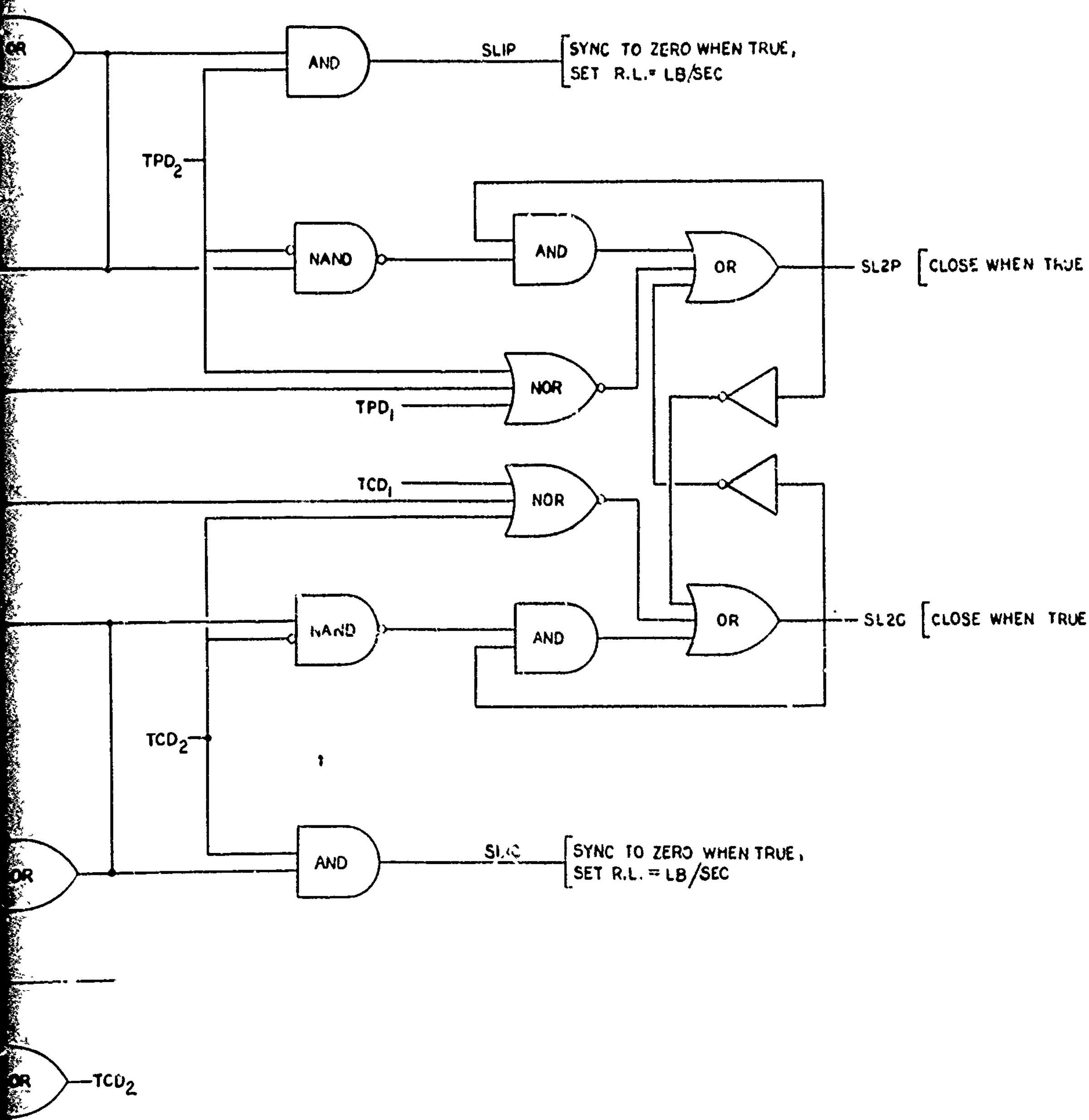


Figure A-9. Control Disengage and Reset Logic.

ROL DISENGAGE AND RESET LOGIC



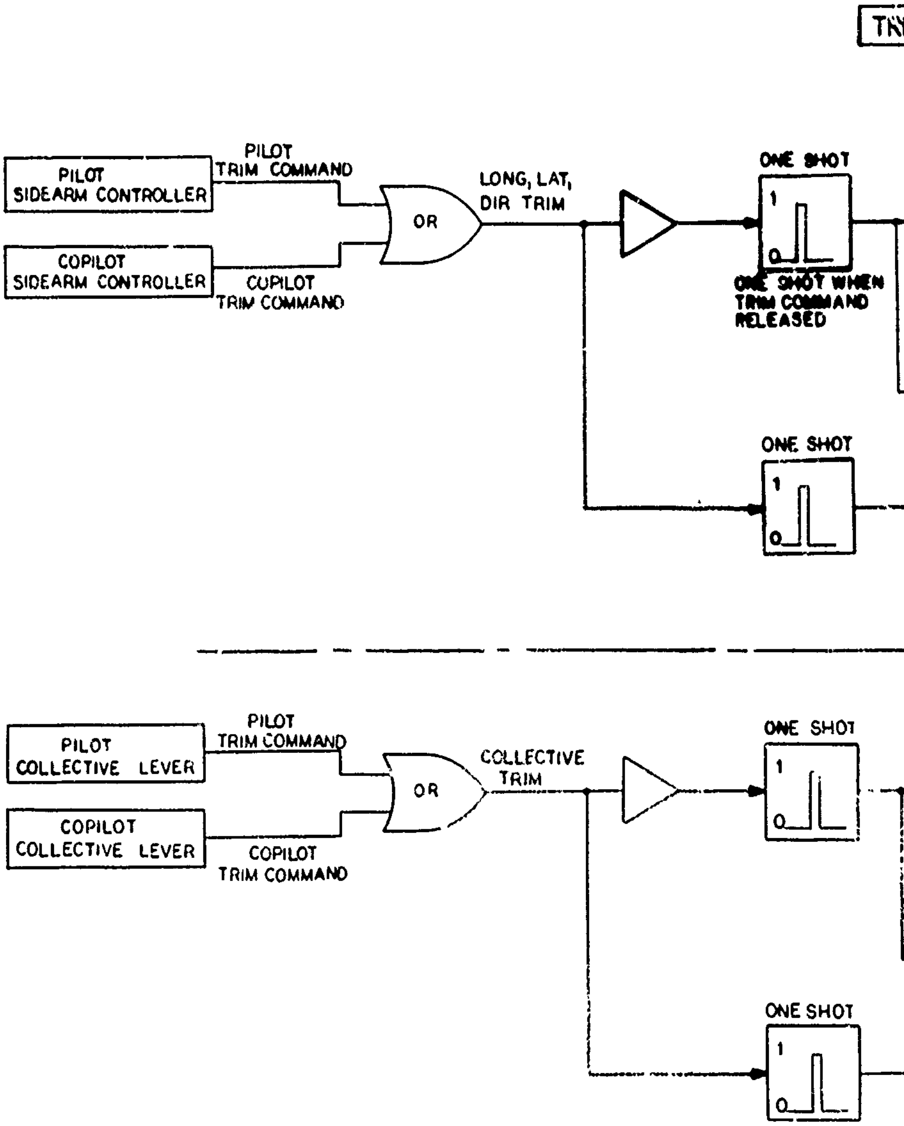
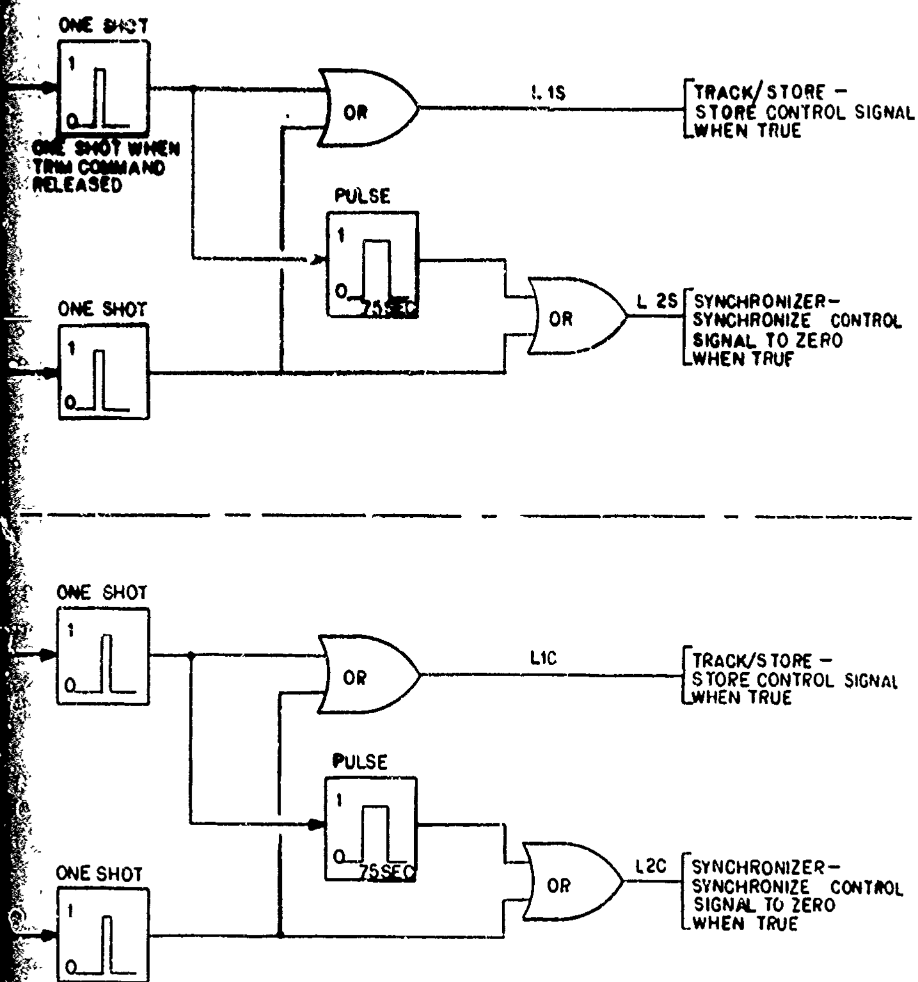


Figure A-10. Trim Command Logic.

TRIM COMMAND LOGIC



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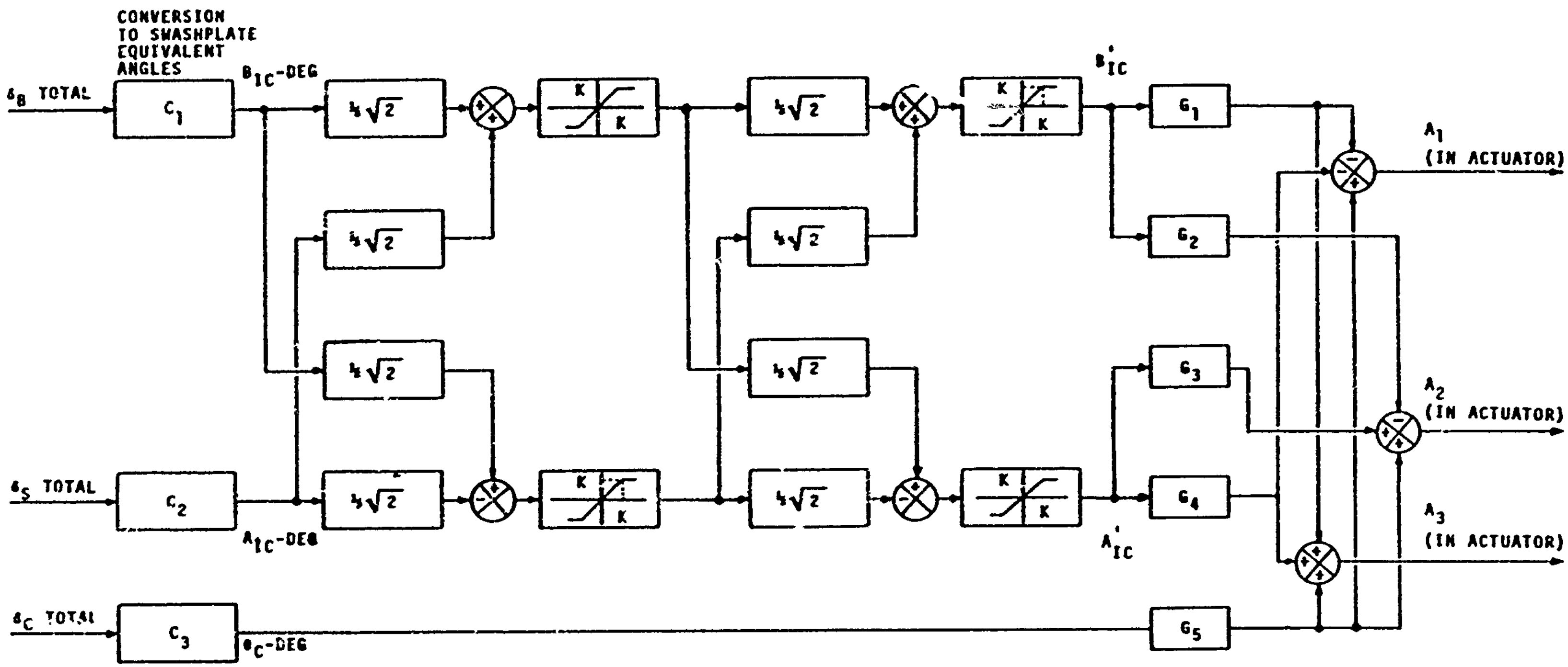

Figure A-11. Cyclic Control Mixing and Cumulative Limiter.

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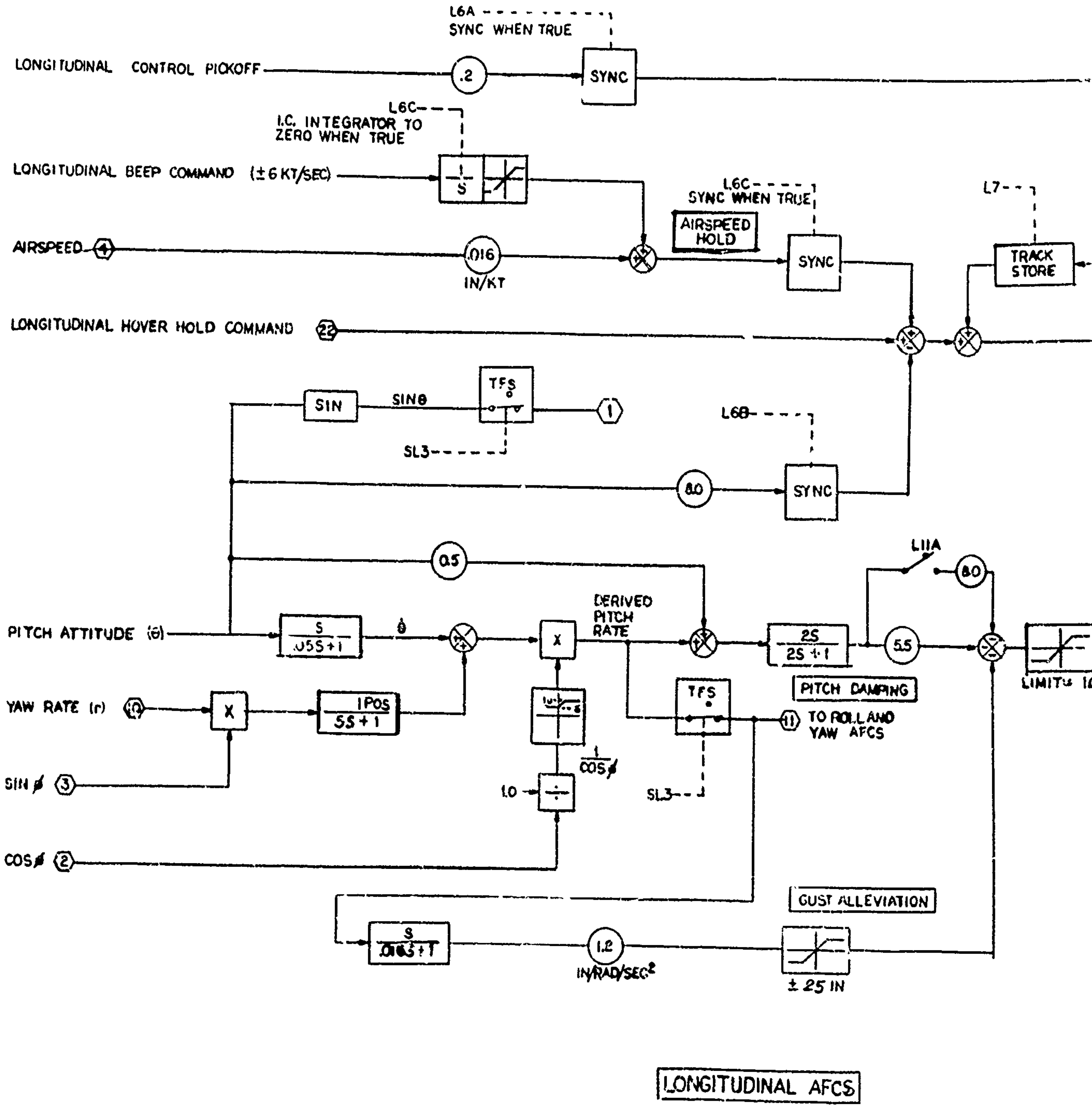
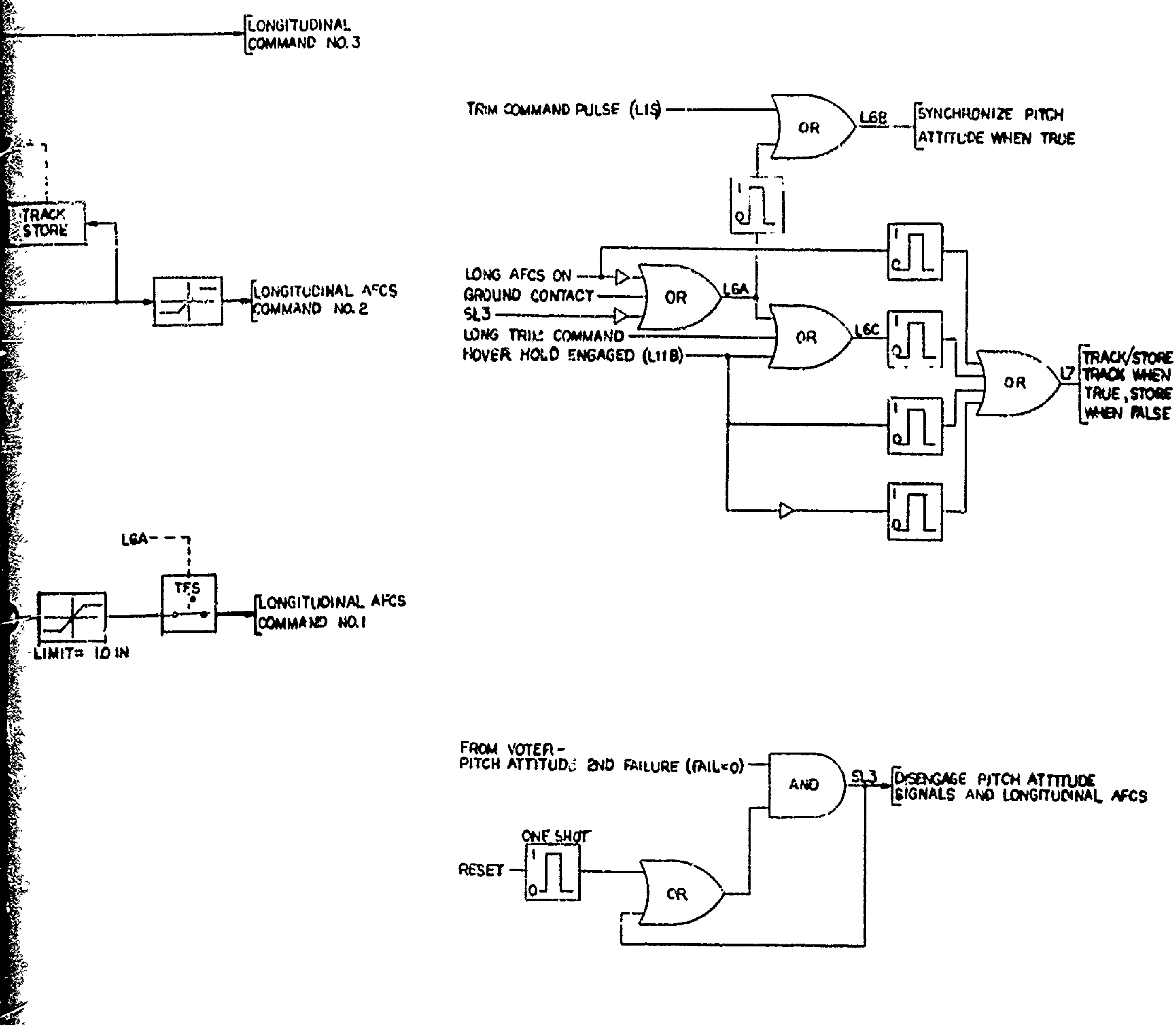


Figure A-12. Longitudinal AFCS Block Diagram.



LONGITUDINAL COMMAND NO. 3
TRACK STORE
LONGITUDINAL AFCS COMMAND NO. 2
L6A
TFS
LIMIT= 1.0 IN
LONGITUDINAL AFCS COMMAND NO. 1
TRIM COMMAND PULSE (L15)
OR
L6B
SYNCHRONIZE PITCH ATTITUDE WHEN TRUE
LONG AFCS ON
GROUND CONTACT
SL3
LONG TRIM COMMAND
HOVER HOLD ENGAGED (L11B)
OR
L6A
OR
L6C
OR
L7
TRACK/STORE TRACK WHEN TRUE, STORE WHEN FALSE
FROM VOTER - PITCH ATTITUDE 2ND FAILURE (FAIL=0)
AND
SL3
DISENGAGE PITCH ATTITUDE SIGNALS AND LONGITUDINAL AFCS
ONE SHOT
RESET
OR

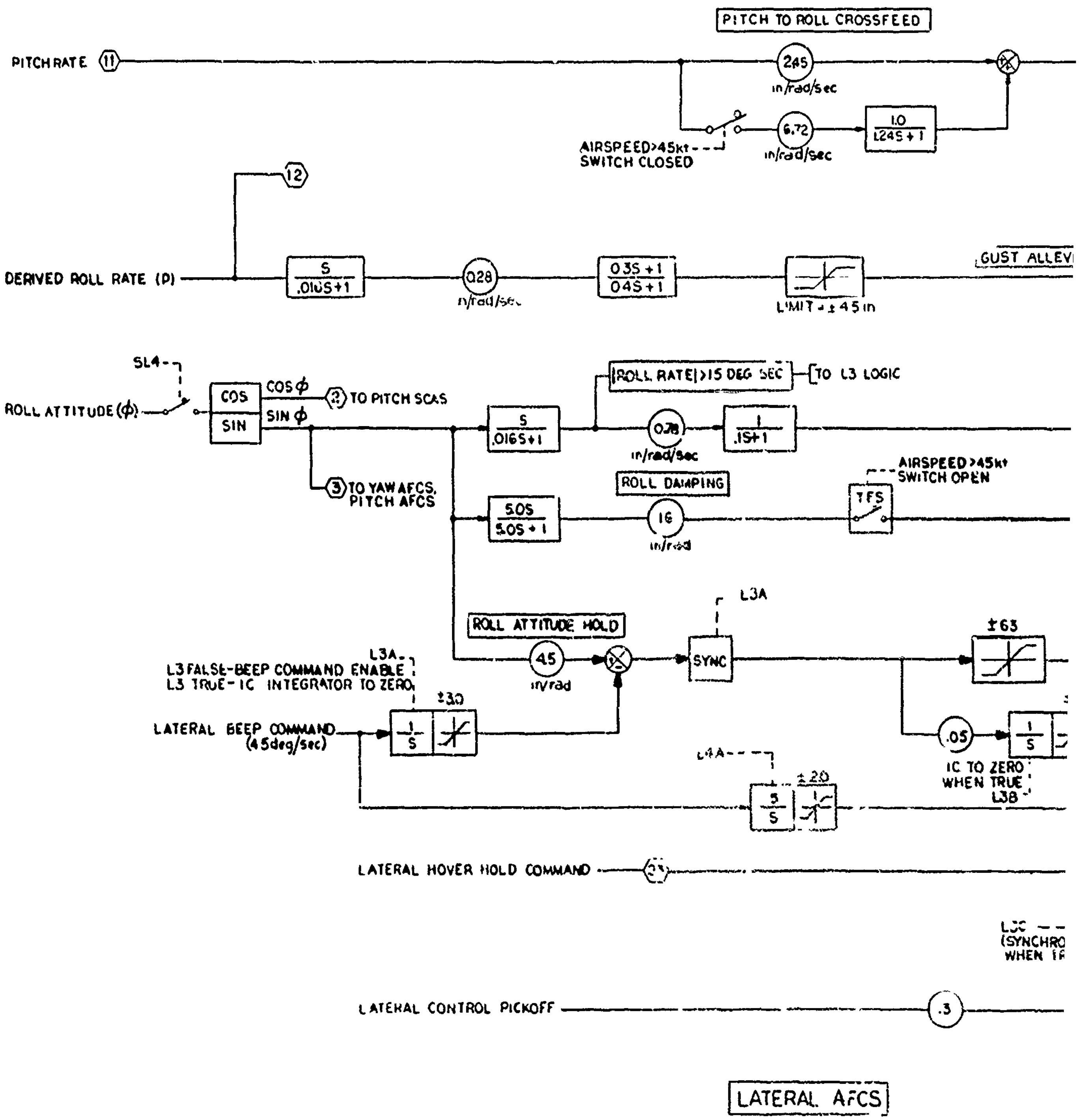
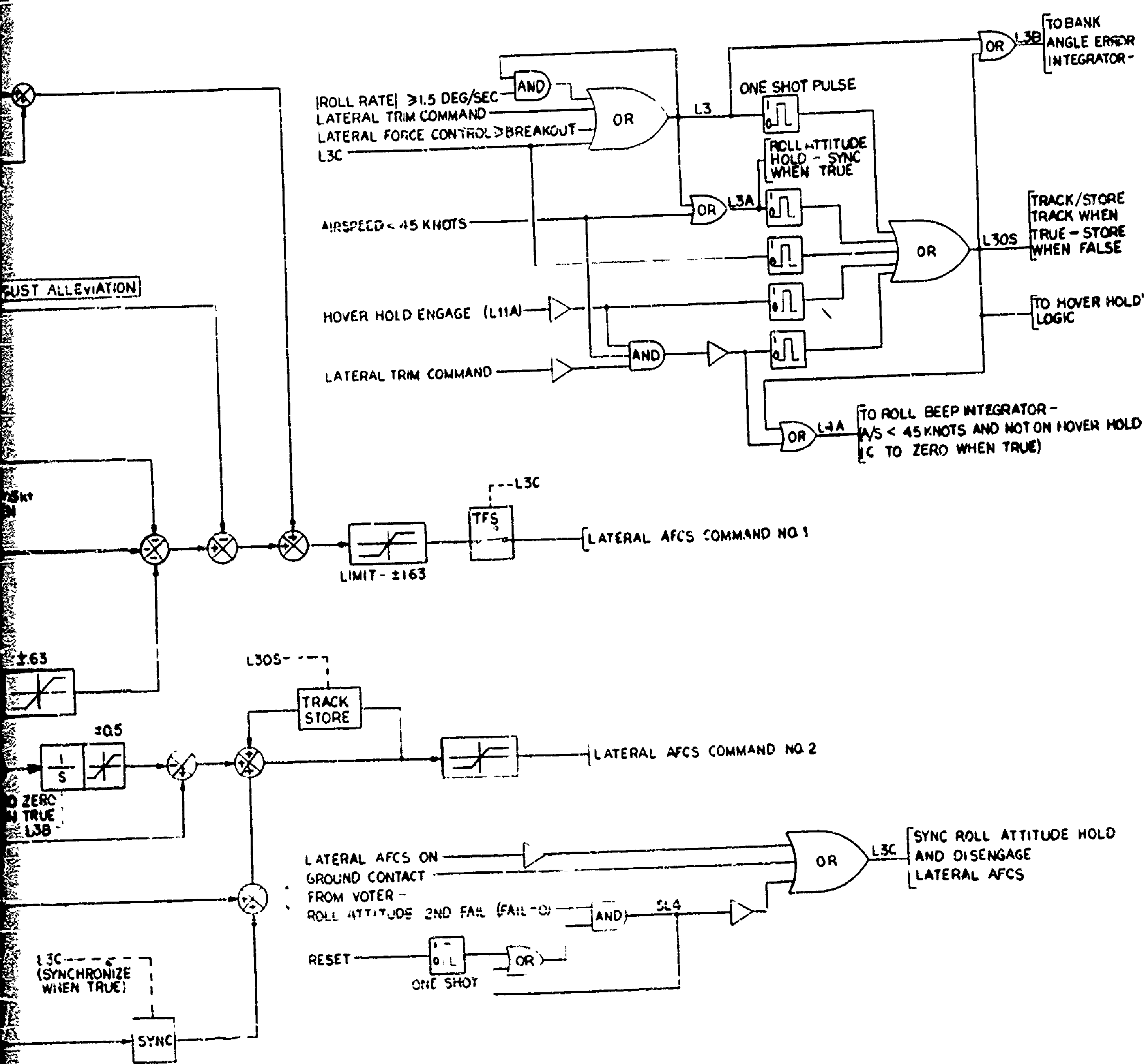


Figure A-13. Lateral AFCS Block Diagram.



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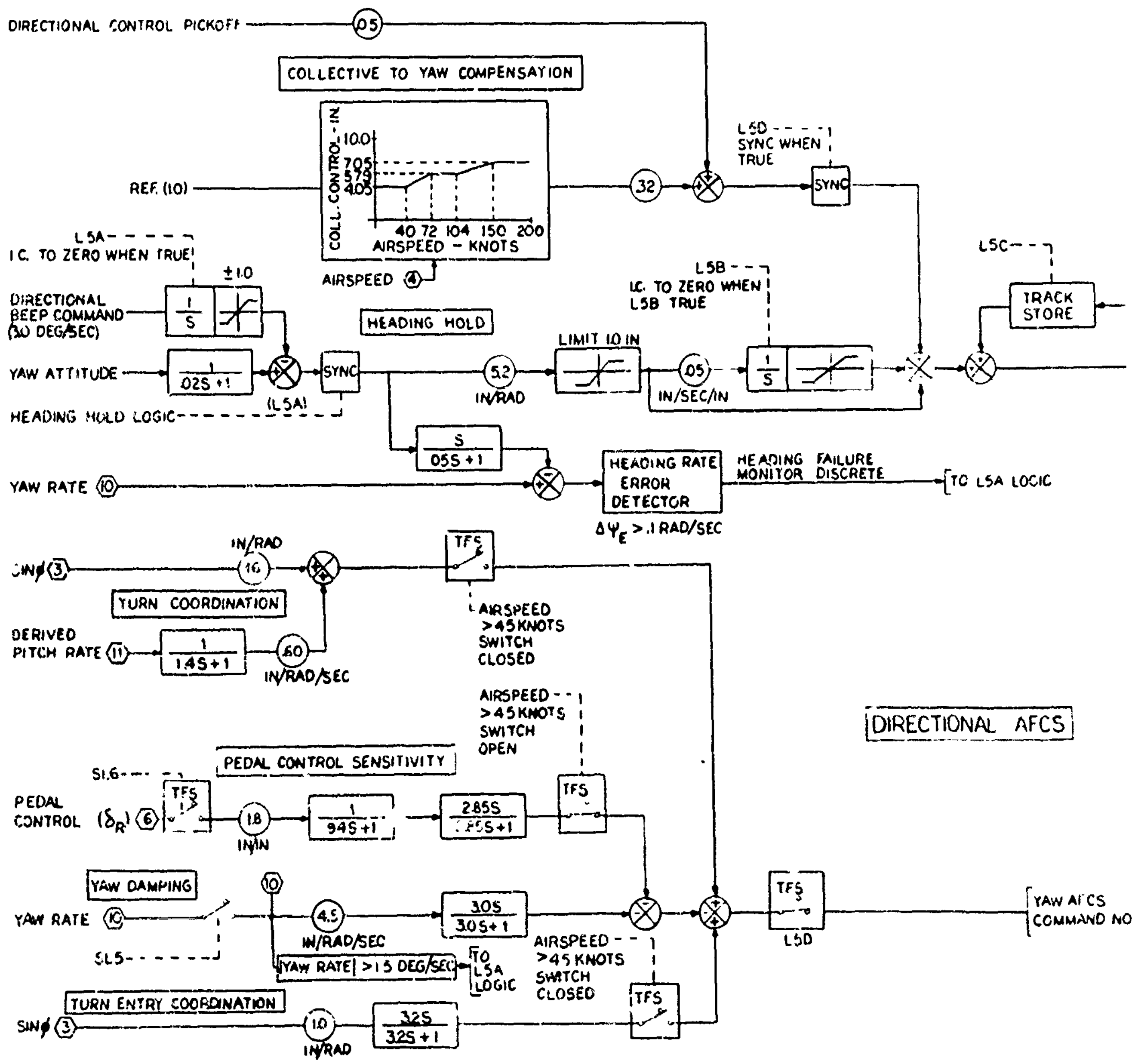

Figure A-14. Directional AFCS Block Diagram.

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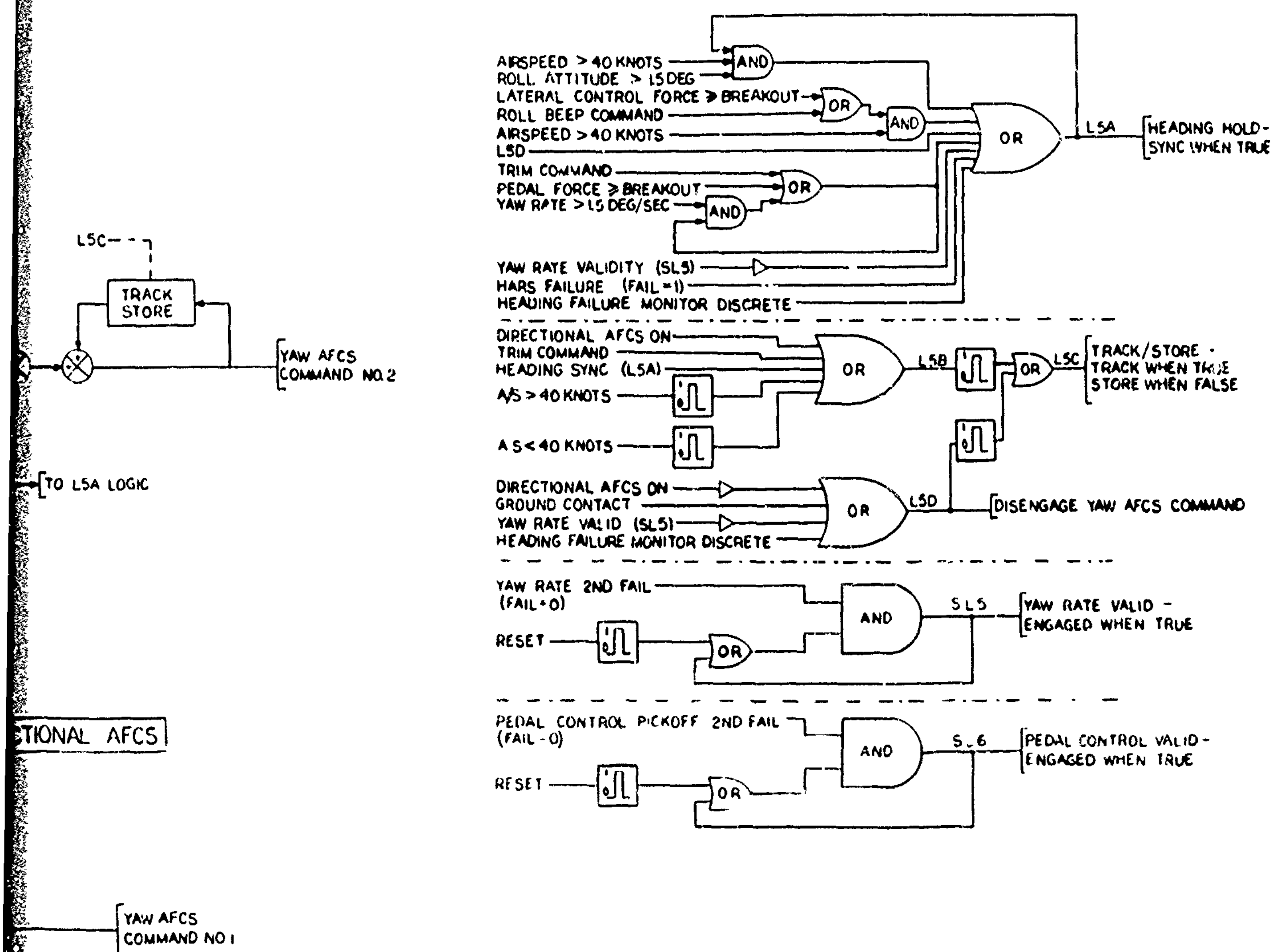

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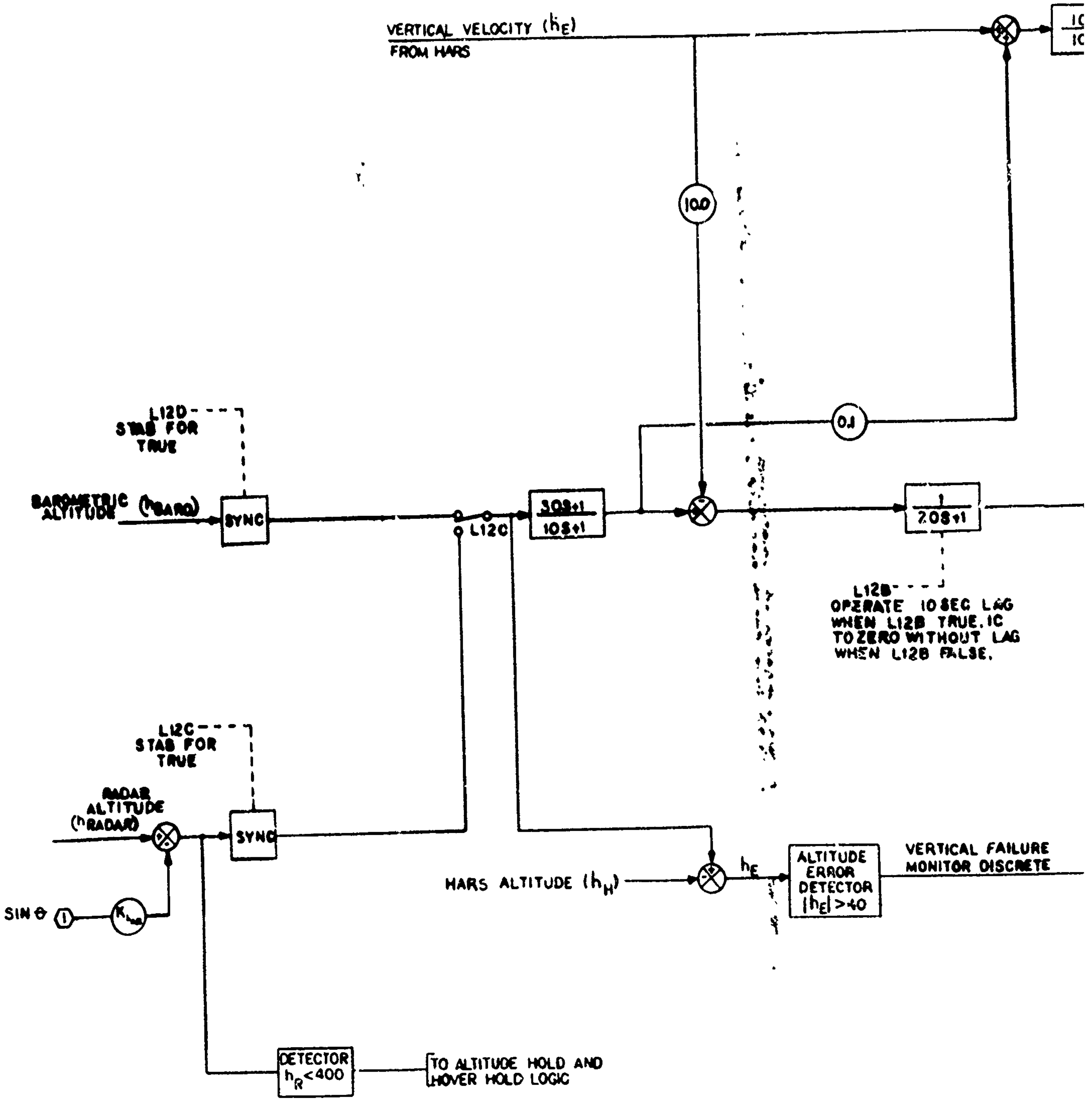
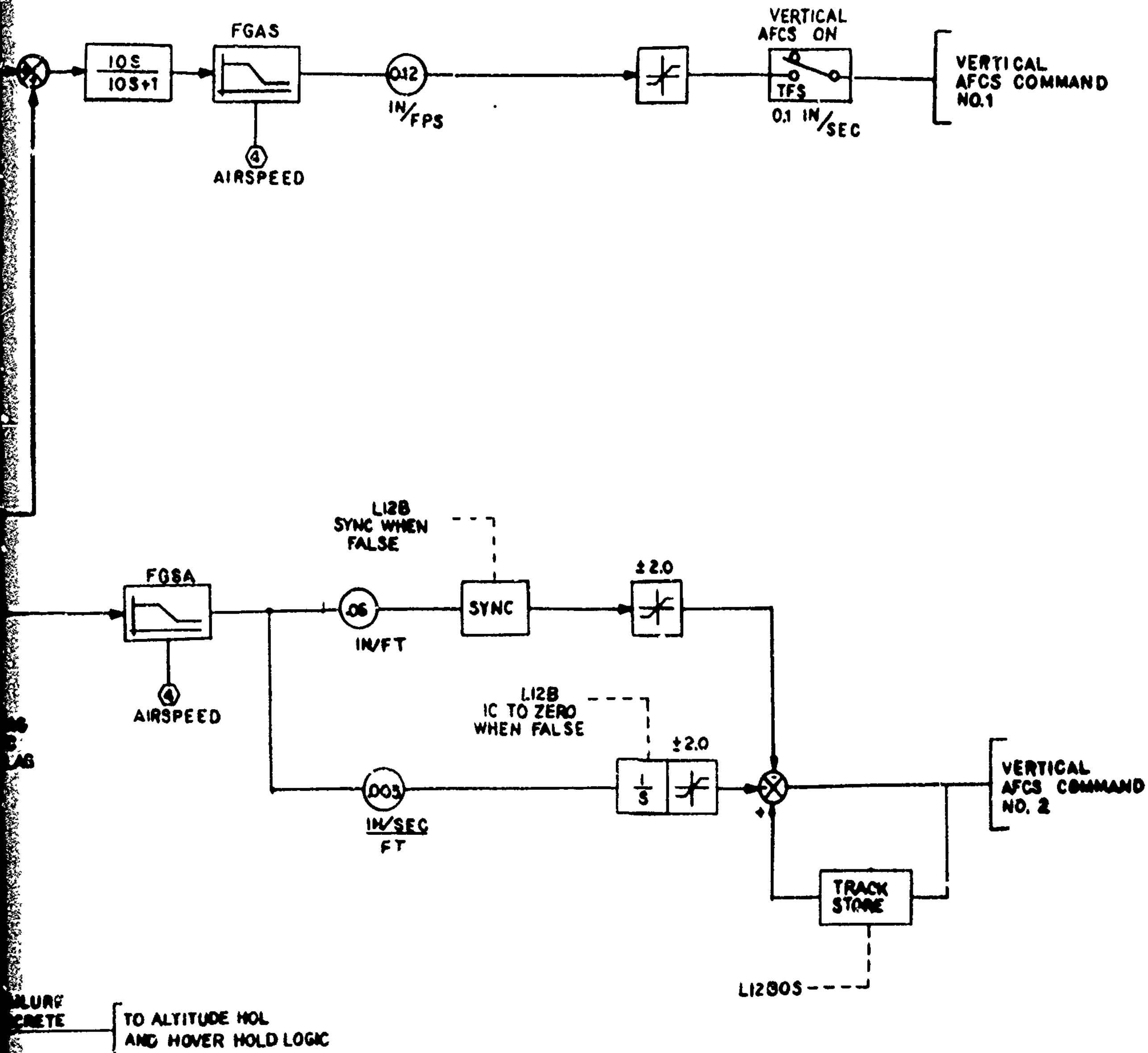


Figure A-15. Vertical AFCS Block Diagram.



ALTITUDE HOLD MODE

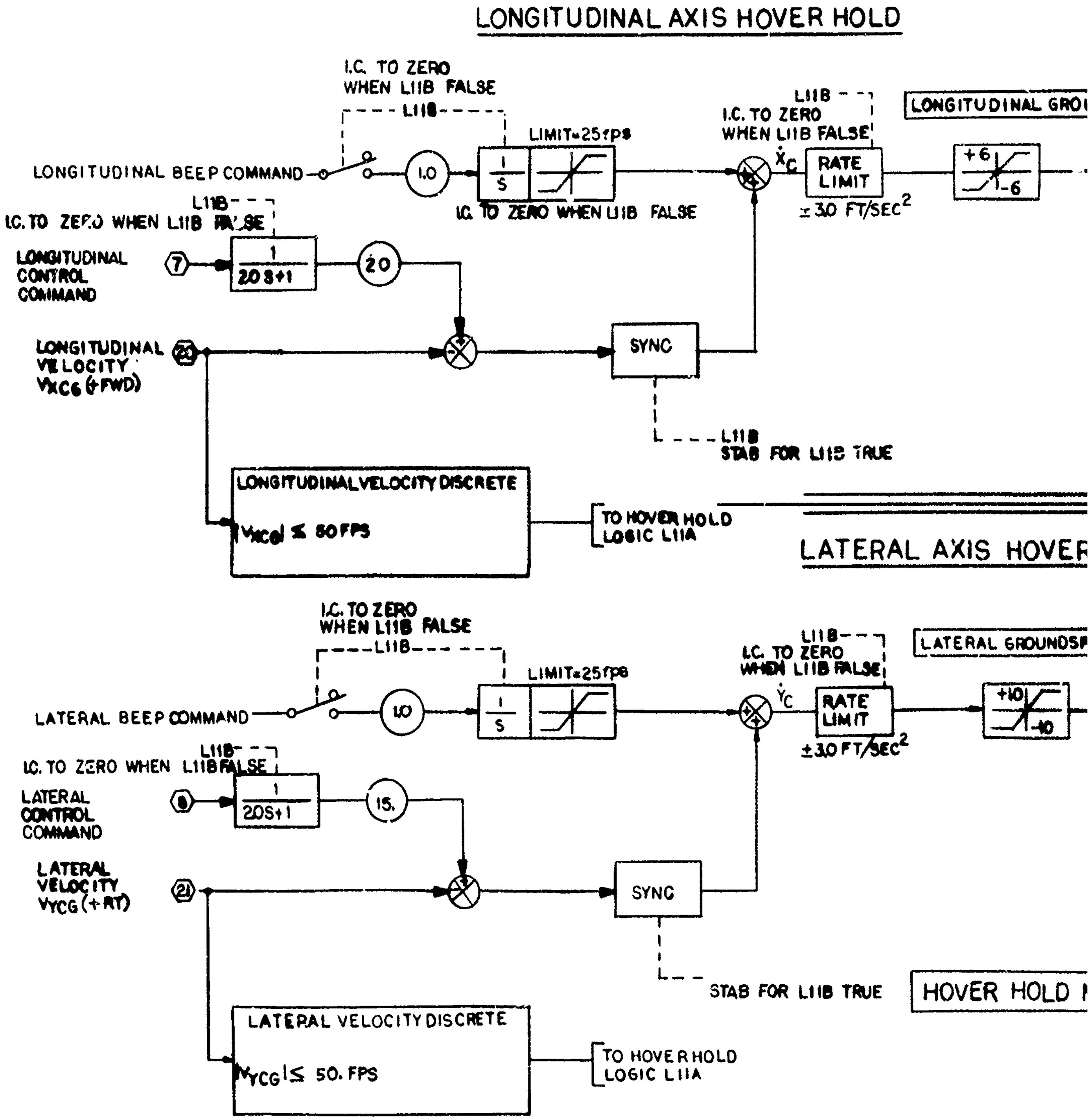


Figure A-16. Hover Hold Mode - Longitudinal and Lateral Axes.

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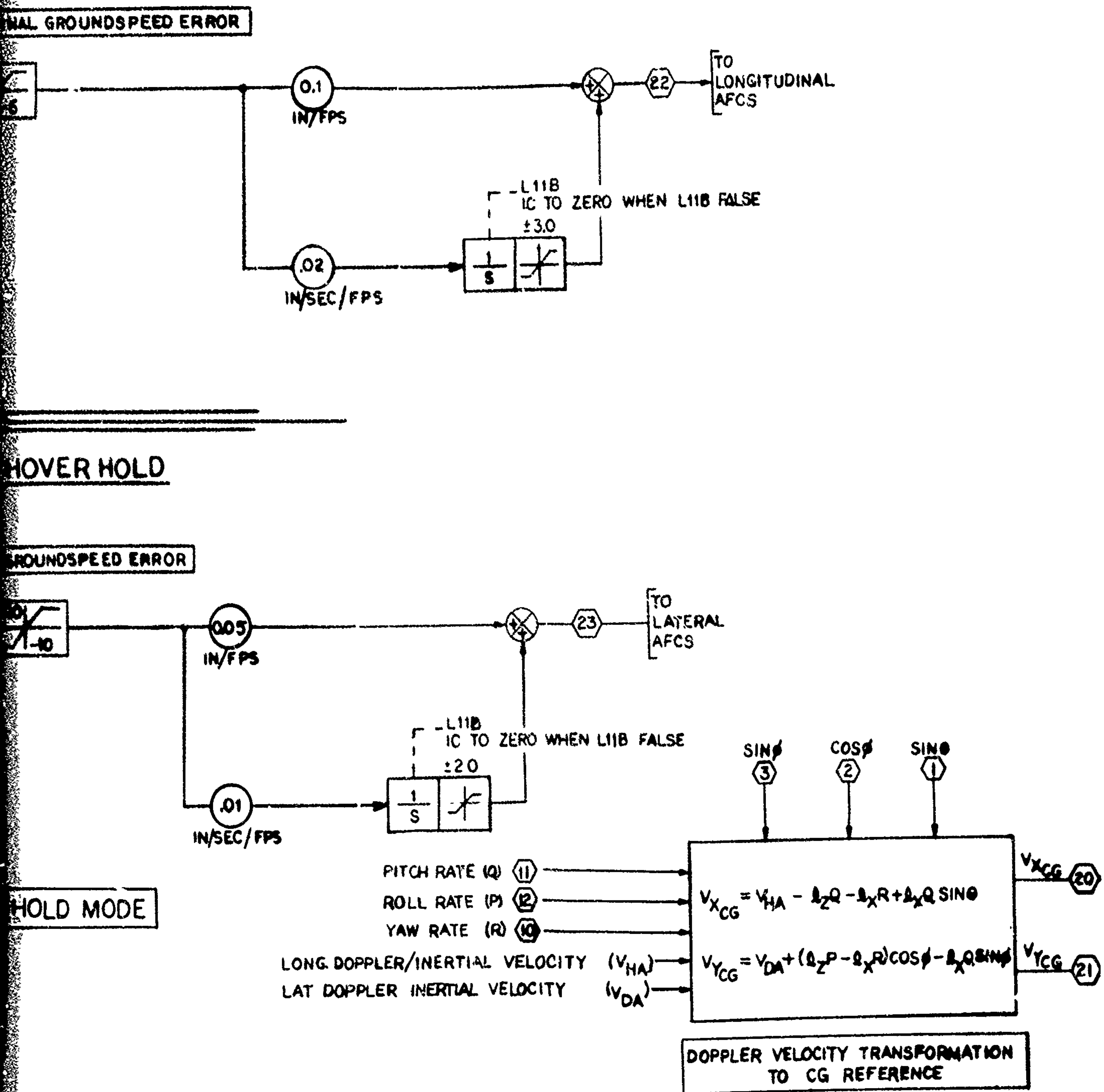


NAL GROUNDSPEED ERROR
0.1
IN/FPS
.02
IN/SEC/FPS
L11B
IC TO ZERO WHEN L11B FALSE
±3.0
1/S
22
TO LONGITUDINAL AFCS
HOVER HOLD
ROUNDSPEED ERROR
10
-10
0.05
IN/FPS
.01
IN/SEC/FPS
L11B
IC TO ZERO WHEN L11B FALSE
±2.0
1/S
23
TO LATERAL AFCS
HOLD MODE
SINϕ 3
COSϕ 2
SINθ 1
PITCH RATE (Q) 11
ROLL RATE (P) 12
YAW RATE (R) 10
LONG. DOPPLER/INERTIAL VELOCITY (V_{HA})
LAT DOPPLER INERTIAL VELOCITY (V_{DA})
$V_{X_{CG}} = V_{HA} - \ell_Z Q - \ell_X R + \ell_X Q \sin\theta$
$V_{Y_{CG}} = V_{DA} + (\ell_Z P - \ell_X R)\cos\phi - \ell_X Q \sin\phi$
$V_{X_{CG}}$ 20
$V_{Y_{CG}}$ 21
DOPPLER VELOCITY TRANSFORMATION TO CG REFERENCE

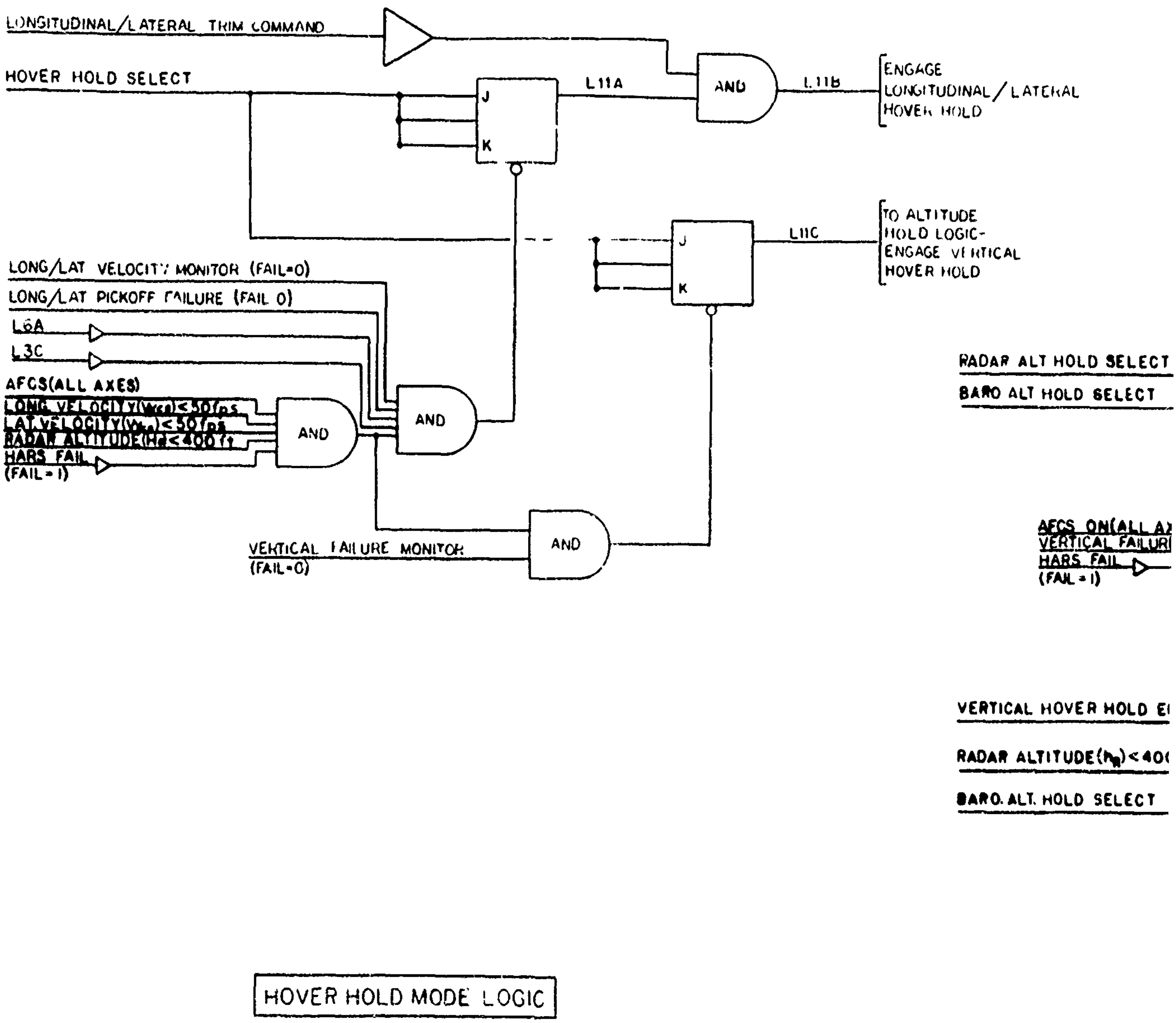
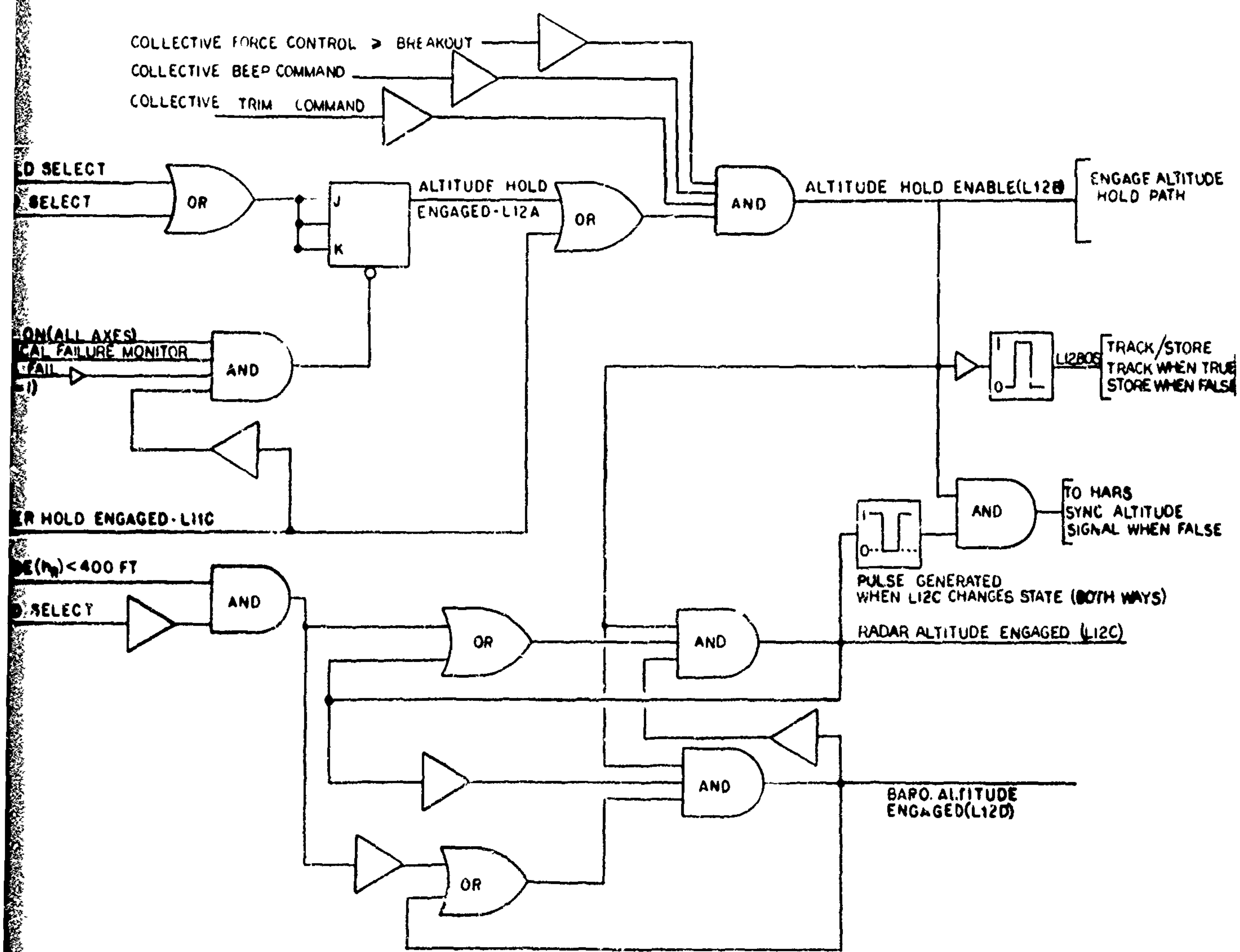


Figure A-17. Hover Hold Mode/Altitude Hold Selectable Mode Logic.



ALTITUDE HOLD MODE LOGIC

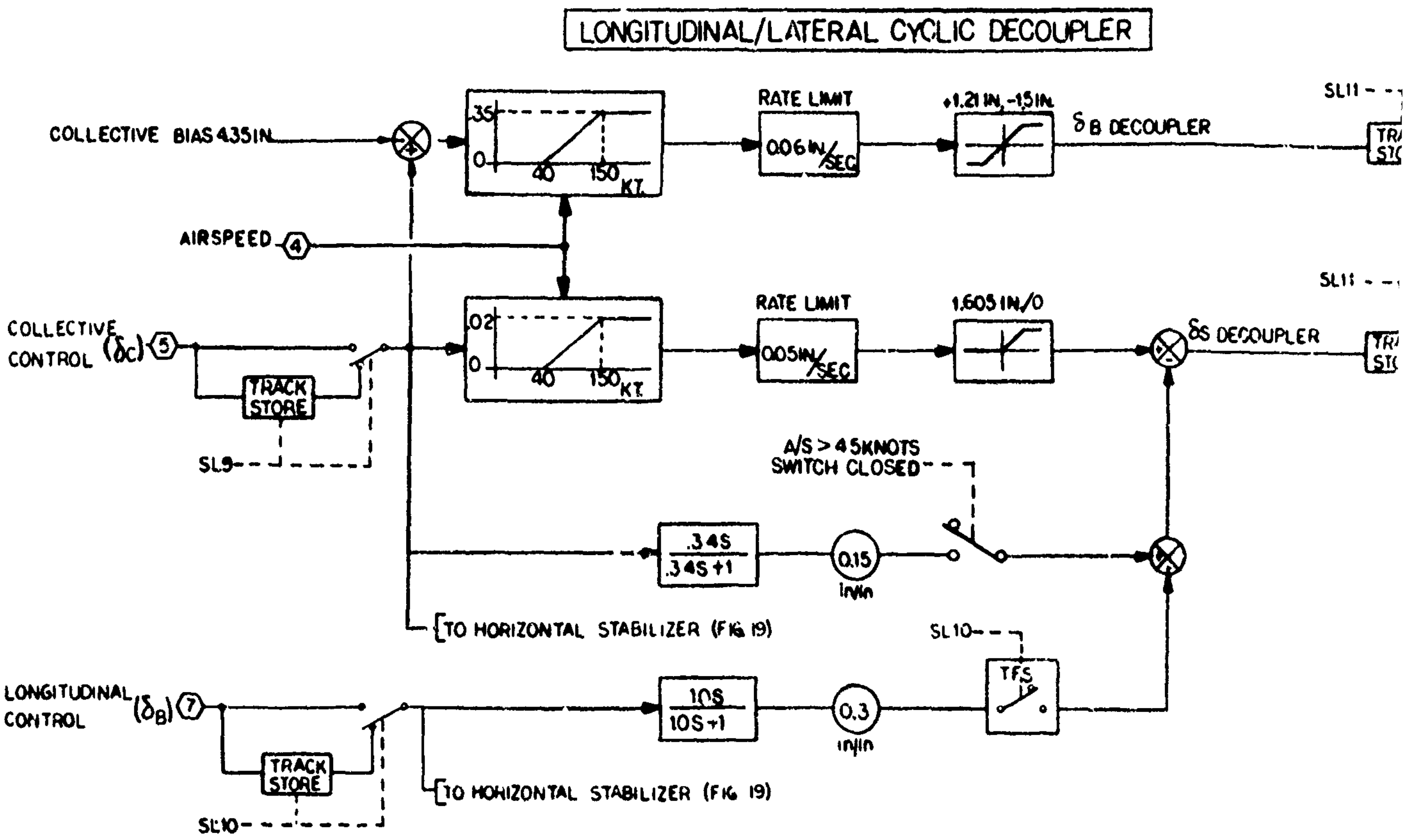
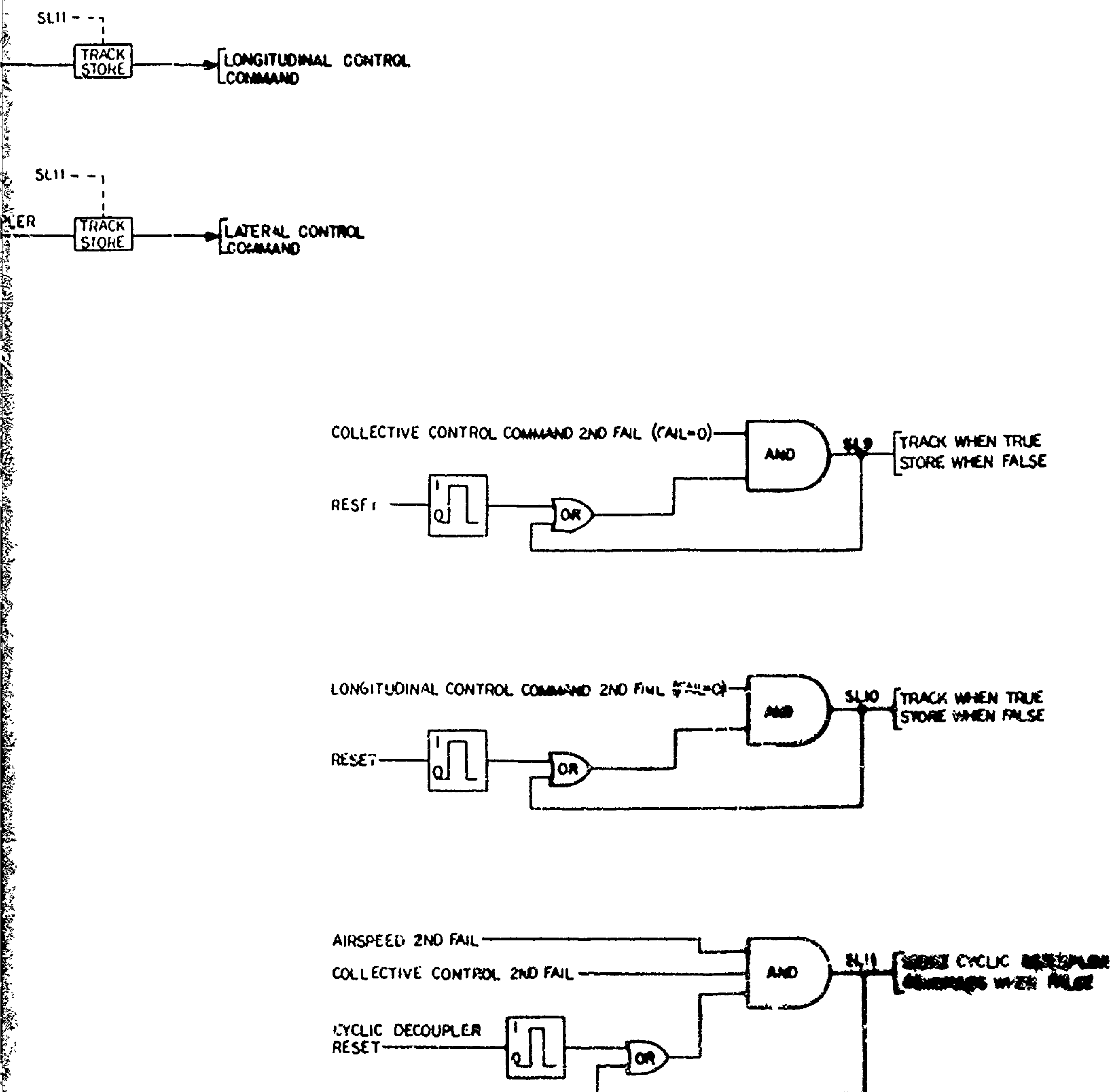


Figure A-18. Longitudinal/Lateral Cyclic Decoupler.



SL11
TRACK STORE
LONGITUDINAL CONTROL COMMAND
SL11
TRACK STORE
LATERAL CONTROL COMMAND
COLLECTIVE CONTROL COMMAND 2ND FAIL (FAIL=0)
AND
SL9
TRACK WHEN TRUE
STORE WHEN FALSE
RESET
OR
LONGITUDINAL CONTROL COMMAND 2ND FAIL
AND
SL10
TRACK WHEN TRUE
STORE WHEN FALSE
RESET
OR
AIRSPEED 2ND FAIL
COLLECTIVE CONTROL 2ND FAIL
CYCLIC DECOUPLER RESET
AND
SL11
CYCLIC
OR

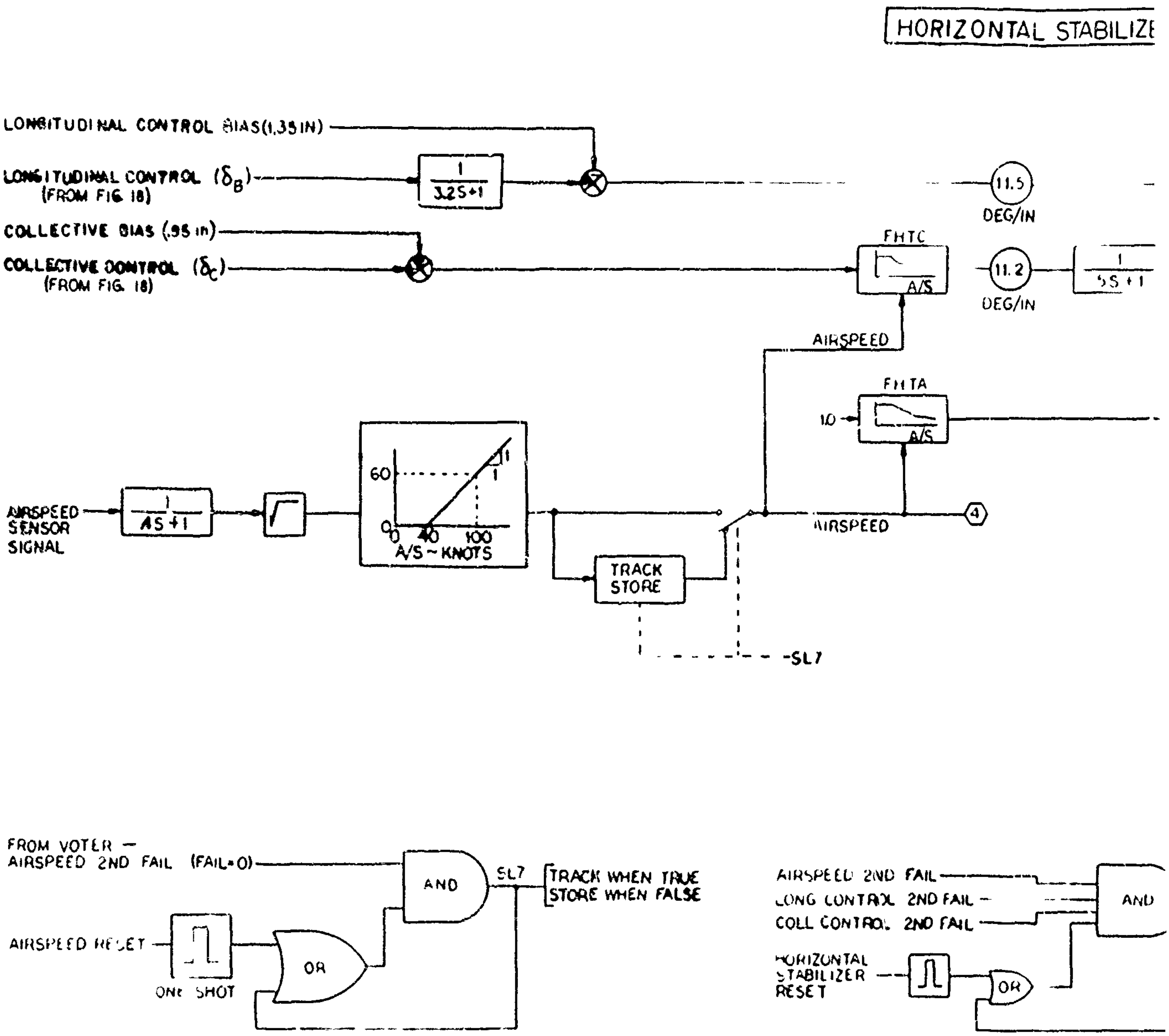
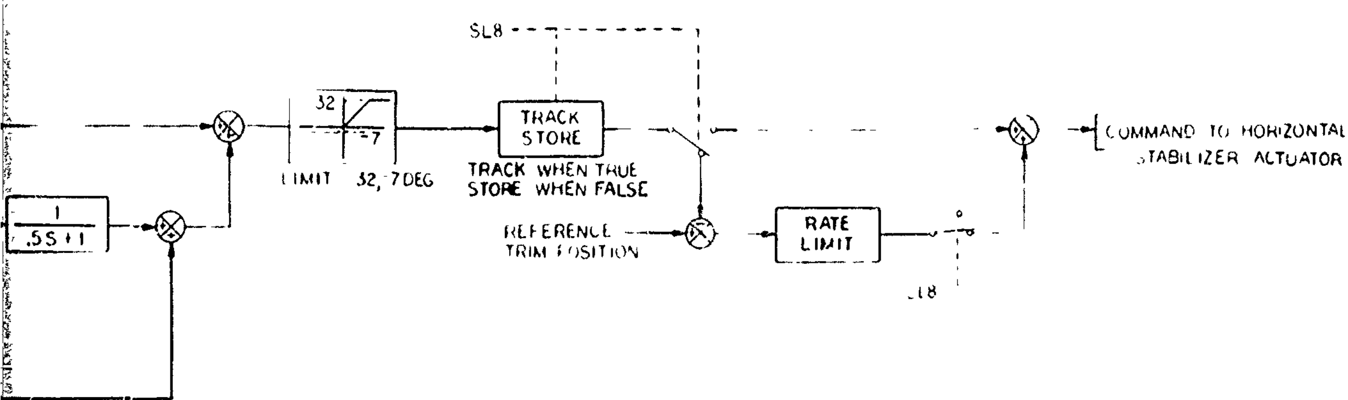
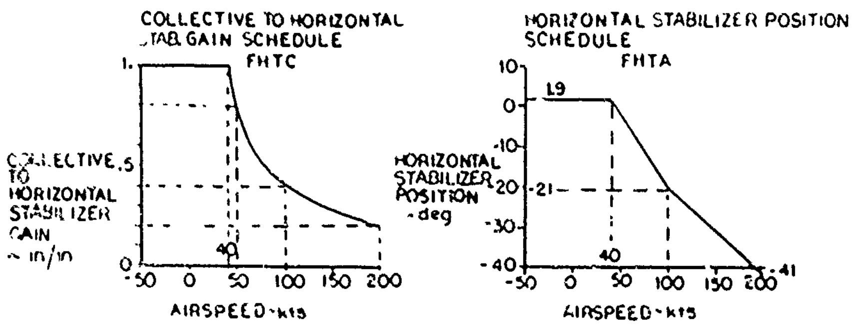
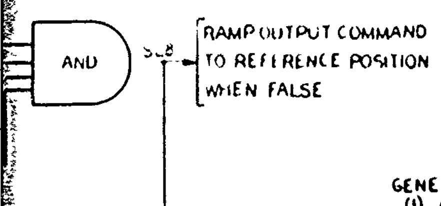


Figure A-19. Horizontal Stabilizer Trim Block Diagram.

TABILIZER TRIM







NOTES APPLICABLE TO SFCS BLOCK DIAGRAMS

GENERAL

(1) ALL AIRSPEED SWITCHES ARE IMPLEMENTED AS PHASED GAINS OVER THE RANGE 45 TO 55 KTS
(2) GAINS ARE EXPRESSED IN EQUIVALENT INCHES OF COCKPIT CONTROL OR DEGRESS OF HORIZONTAL STABILIZER MOTION
(3) ALL SWITCHES ARE SHOWN IN THE FALSE POSITION

There shall be triplex sensors to each processor unit. These redundant sensor inputs shall be voted before going into the AFCS (Figure A-4). For first failure, the failed signal shall be rejected and the remaining two shall be averaged.

For second failure, actions shall be taken as listed in Table A-1. In addition, there shall be self-monitored nonredundant signals coming into the AFCS via dedicated optical links (Figure A-4). Actions to be taken on failure are listed in Table A-2.

- Outputs

Outputs from three AFCS channels shall be voted by axes (longitudinal, lateral, directional, and collective). Only the failed axis shall be shut down. All axes will be shut down for computer failure. For first failure, the failed signal shall be rejected; for second failure, the signals shall be ramped to a reference value.

The output of the voter shall be passed through the authority/rate limit network shown in Figure A-20.

BITE Fault Isolation

The BITE, in conjunction with the failure detection circuits in each channel, shall provide the following functions for either preflight, inflight, or periodic checkout:

1. Conduct a test that will detect any channel failures.

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2. Verify the integrity of the failure detection circuits, such as:
   a. Primary system: actuator current monitor, control stage position with current comparison, clock monitor, parity check, PFCS/AFCS interface voter.
   b. Automatic system: sensor voter, signal sensor monitor.
3. Provide the capability of isolating any fault to a line replaceable unit (LRU) with a 90-percent probability of correct diagnosis.

Display/Control

The following control functions shall be provided:

1. Display of failure indication as detected by the failure detection circuits locating the failure to the failed channel and LRU.
2. Display of the BITE checkout results.

The following control functions shall be provided:

1. Capability to reset the failed channel.
2. Transmission of test signals for initiation of BITE automatic checkout sequence

TABLE A-1. ACTION ON SECOND FAILURE AT SENSOR INPUTS

Sensor	Action on second failure.
Airspeed	1. Store last valid value of airspeed. A reset shall be available. This will accomplish the following tasks: - Longitudinal AFCS (Figure A-12): Freeze airspeed hold input. - Lateral AFCS (Figure A-13): Store in last valid mode (keep switches in last valid positions). - Yaw AFCS (Figure A-14): Store in last valid mode. - Longitudinal/lateral cyclic decoupler (Figure A-17): Store last valid signal. - Yaw compensation (Figure A-18): Store last valid signal. 2. Ramp horizontal tail command (Figure A-19) to a reference value.
Pitch Attitude (Figure A-12)	These actions shall be taken. A reset shall be available. 1. Turn off longitudinal AFCS command No. 1. 2. Store longitudinal AFCS commands No. 2 and 3 3. Ramp: Pitch signals to other axes to zero.
Roll Attitude (Figure A-13)	These actions shall be taken. A reset shall be available. 1. Shut down lateral AFCS command No. 1 2. Store lateral AFCS command No.2 3. Ramp: Roll signals to other axes to zero.
Yaw Attitude (Figure A-14)	These actions shall be taken. A reset shall be available. 1. Shut down yaw AFCS command No. 1 2. Store yaw AFCS command No. 2 3. Ramp: Yaw signals to other axes to zero.
Pickoff	These actions shall be taken. A reset shall be available. 1. Yaw axis - δ_R pickoff (Figure A-14): Ramp pickoff signal to zero.

TABLE A-1. (Continued)

Sensor	Action on second failure.
Pickoff	2. Hover hold - Longitudinal/lateral pickoff (Figure A-16): Disengage longitudinal/lateral hover hold 3. Cyclic decoupler and yaw compensator - δ_C pickoff (Figure A-18): Store last value. 4. Horizontal tail - δ_B and δ_C pickoff (Figure A-19): Ramp horizontal tail command to some reference value.

TABLE A-2. ACTION ON FAILURE OF NONREDUNDANT SENSORS

Mode	Action
Vertical Hold (Figures A-15 and A-17)	The vertical position from the HARS shall be compared with the radar altitude (h_R) if h_R <40 feet, and with the barometric altitude (h_{BARO}) if h_R >400 feet. The vertical hold mode shall be disengaged in case of failure. The mode shall be reset by pilots' engage command only if failure has been cleared.
Hover Hold (Figures A-16 and A-17)	The doppler longitudinal/lateral velocities are monitored by the computer inside the HARS. The failure signal from the HARS shall be used to disengage the hover hold mode. The mode shall be reset by pilots' engage command only if failure has been cleared. The first word from the HARS serial digital output is the status word which includes: 1. HARS detected failure. 2. Digital attitude available/valid. 3. Align status. 4. Maneuver condition - excessive rate. 5. Doppler/inertial longitudinal/lateral velocity comparison.

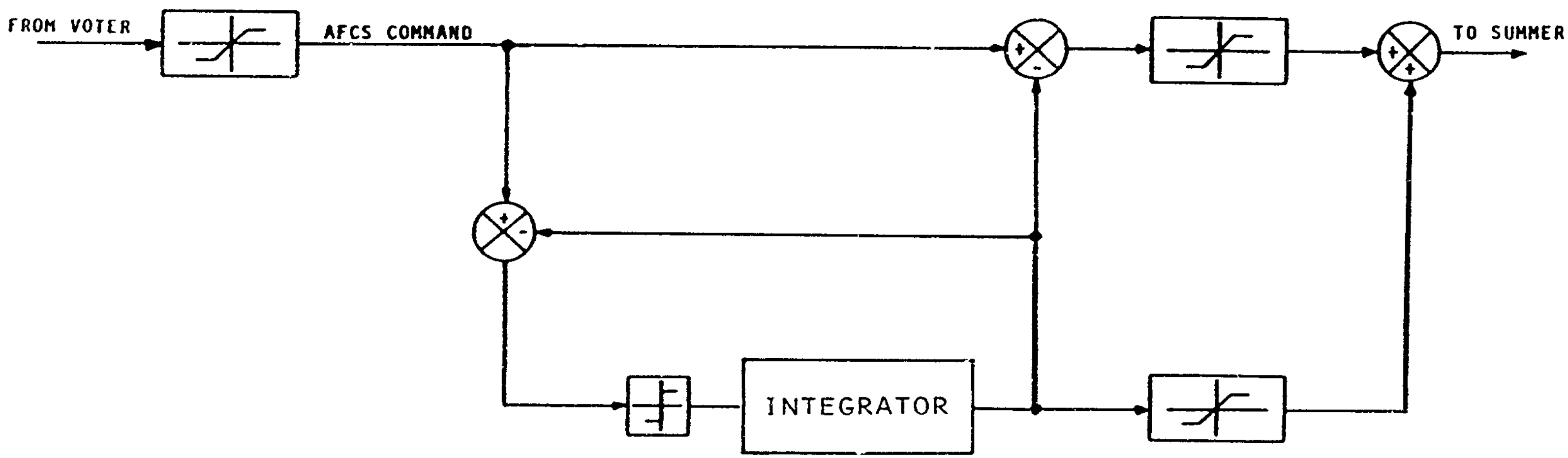

Figure A-20. AFCS Authority/Rate Limit Network (Typical for Each Axis).

Rotor Control Actuators

There shall be three electrohydraulic actuators for the main rotors and one electrohydraulic actuator for the tail rotor. The tail rotor control actuator is identical to the main rotor actuators except for output load and stroke requirement. The envelope for the actuator shall not exceed the dimensions given in Figure A-21.

- Servo Loop

Figure A-5 shows the rotor control actuator servo loop. When the actuator is at rest, the control stage piston and transducer are null; the power stage position transducer matches the actuator command. When the actuator command changes, the control stage piston assumes a position proportional to the generated error. This causes the power stage valve and piston movement that reduces the servoamplifier error to zero, and the actuator is again at rest. The two-stage design effectively decouples control and power stages so that the redundancy management can be handled in the control stage where the rotor loads are not reflected and cannot upset the redundancy management.

- Function Description

Figure A-6 shows the rotor control actuator. Servoamplifier current positions the jet pipe of the single stage electro-hydraulic valve (EHV), which produces pressure and flow proportional to current. The EHV flow moves the control stage piston, which, in turn, positions the power valve via an anti-jam bungee. Power stage output velocity is proportional to power stage valve
27 F position. Either linear variable differential transformers (LVDT) or optical transducers shall be used to measure power and control stage piston positions. These transducers close the loops as discussed in the previous paragraph. A separate control stage position transducer shall be provided for each control stage. Control stage hydromechanical failures shall be detected by comparison of the control stage position with EHV current. Sufficient overtravel is provided in the control stage piston and anti-jam bungee to allow full recovery from a control stage jam at the full power stage valve displacement. The only change under these conditions will be a 50-percent reduction in control stage loop gain. The system will be designed to tolerate this condition with no change in power loop stability.

- Anti-jam Power Stage Servo Valves

The actuator shall be designed to incorporate two independent anti-jam hydromechanical control valves, one for each section of the dual power control actuator. Each valve shall provide for mechanical sensing of primary spool jams and shall be designed to incorporate a manual reset to aid in trouble-shooting. Each anti-jam valve shall, in addition, provide a means for remote indication as well as ground checkout capability.

Means shall be provided within the package to shut off the supply pressures to the anti-jam valve in the event a jam is sensed. The jam shutoff mechanism shall be capable of allowing the control valve to return to normal operation in the event a jam is cleared, by cycling the system supply pressure.

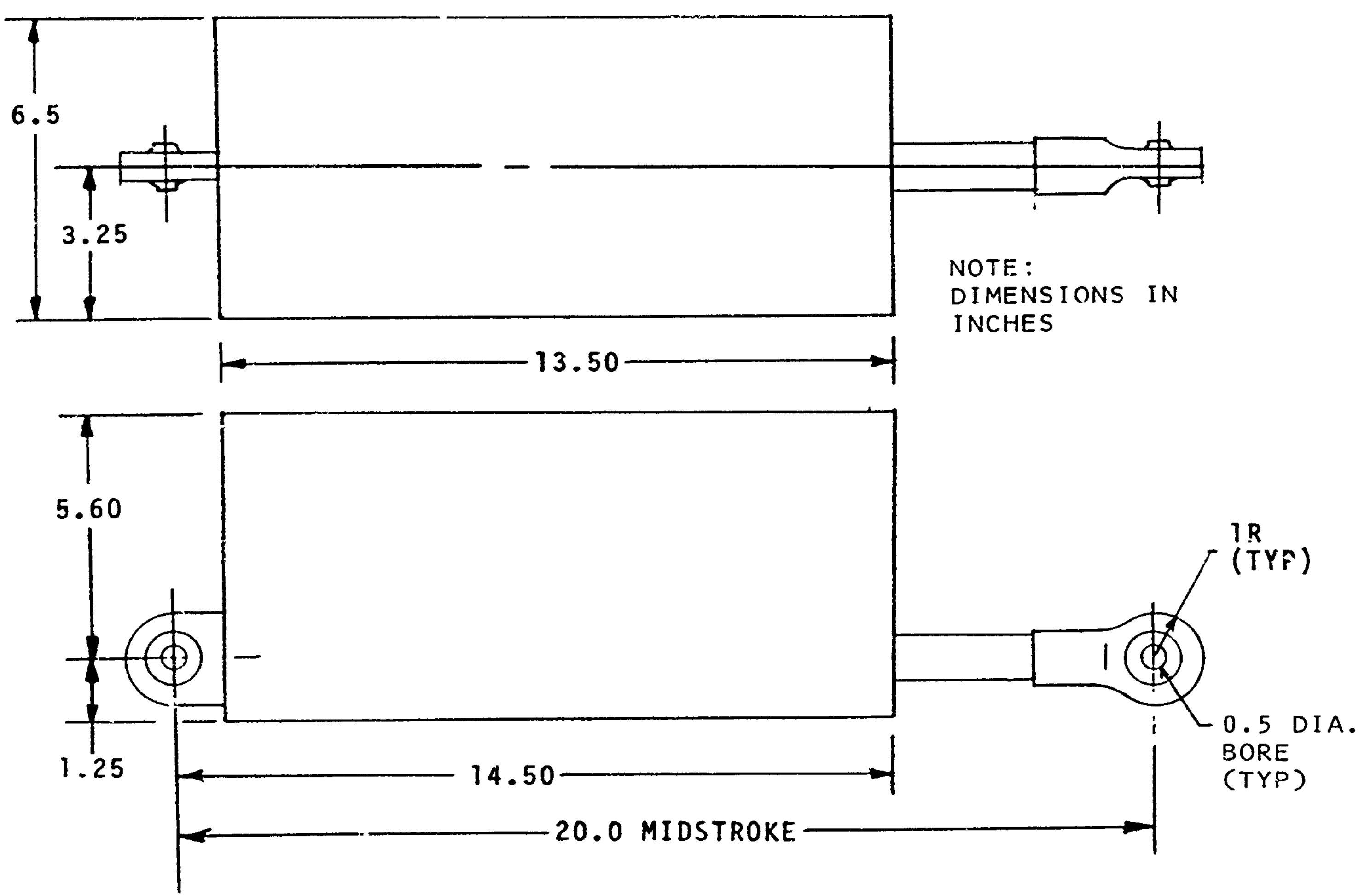

Figure A-21. Rotor Control Actuator Envelope.

The package shall, in addition, incorporate a suitable pressure-operated bypass valve that will allow the appropriate cylinder sections of the package to be interconnected with the system return fluid both during operation with a single jammed primary spool and during loss or shutoff of system supply pressure.

If pressure switches are used as a means of providing remote indications of valve jams, the switch design and testing shall be in accordance with MIL-S-8932 (Reference 11). The operating voltage shall be 18-30 VDC with a current rating of 2.5 amperes maximum at 28 VDC. Switch settings shall be as follows, if applicable:

1. Actuate with increasing pressure – 2400 psi maximum.

2. Actuate with decreasing pressure – 1500 ±100 psi.

- Interface

Figure A-6 shows the interface between the rotor control actuator, the flight control processor unit, the electrical and hydraulic supplies, and the mechanical output. Each of the flight control processor units shall provide an electrical current signal to each electrohydraulic valve. The LVDTs or the optical transducers will provide information to their respective control unit about the actuator piston position, servovalves, and differential pressure sensor.

- Power Actuator Arrangement

Figure A-22 depicts the triple power actuator arrangement, which eliminates the eccentric load condition when one system is inactive.

- Actuator Performance

Table A-3 shows the performance requirements for the actuators.

Horizontal Stabilizer Control

Figure A-23 is a block diagram of the horizontal stabilizer actuator and control circuitry. This arrangement shall provide fail operative/fail off control of the stabilizer via a torque-summing electromechanical actuator. Command signals from the AFCS computer and actuator position signals shall be voted to select the drive signal for the motors. Offsets in the motor outputs shall be equalized by adjustment of motor current. Equalization current shall be limited and sampled to detect amplifier/motor failures and to shut down the failed unit. On second failure, the pilot may reengage and attempt to command the stabilizer to a trim position. The motors used in the actuator shall be of a brushless type. Output gearing shall be of a low friction type to permit back driving of motor sections that are shut down. A dual low-friction screw jack shall be used for the output. A brake with coils, operated by each channel, shall be interfaced with the gearing and shutdown circuitry so that it is on when power is off of all motors.

11. Military Specification, MIL-S-8932, SWITCH, PRESSURE, AIRCRAFT, GENERAL SPECIFICATION FOR, Department of Defense, Washington, D.C., 28 January 1965.

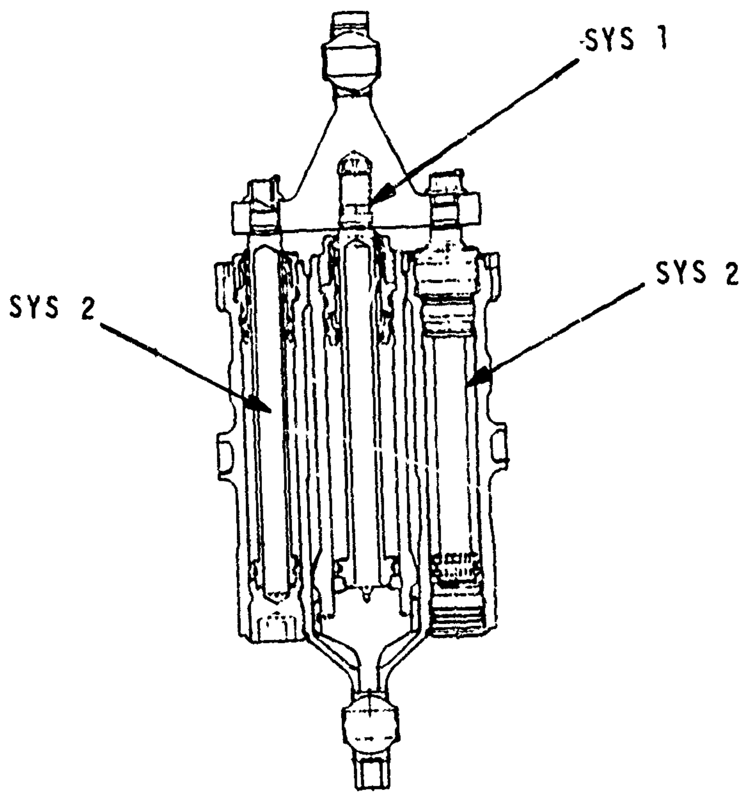

Figure A-22. Power Stage Cylinder Arrangement.

TABLE A-3. ACTUATOR PERFORMANCE

ACTUATOR	SPECIFICATION
Main Rotor	Stroke. ± 2.83 in. measured at actuator output point No load velocity (dual): 3.9 in./sec with 2800 psi delta P Loaded velocity (dual): 2.0 in./sec against 1260 lb @ 2800 psi delta P (single): 3.0 in./sec against 473 lb @ 280 psi delta P Fatigue (tensile): 450 ± 1350 lb (comp.): 2250 ± 1350 lb Limit (tensile and comp.): 8100 lb @ 4500 psi delta P Stall (dual): 5400 lb @ 3000 psi delta P (single): 2700 lb @ 3000 psi delta P
Tail Rotor	Stroke: ± 1.000 in. measured at actuator output point No load velocity (dual): 1.32 in./sec Loaded velocity (dual): 0.66 in./sec against 1500 lb @ 3000 psi delta P Fatigue (tensile or comp.): 535 ± 23 lb for steel parts (infinite life) Stall (dual): 2100 ± 100 lb min. @ 3000 psi delta P (tensile or comp.)
Horizontal Stabilizer	Full Stroke: 3.25 in., 53° of Stabilizer Angular Motion Loaded Velocity (dual): 0.58 in./sec against 750 lb

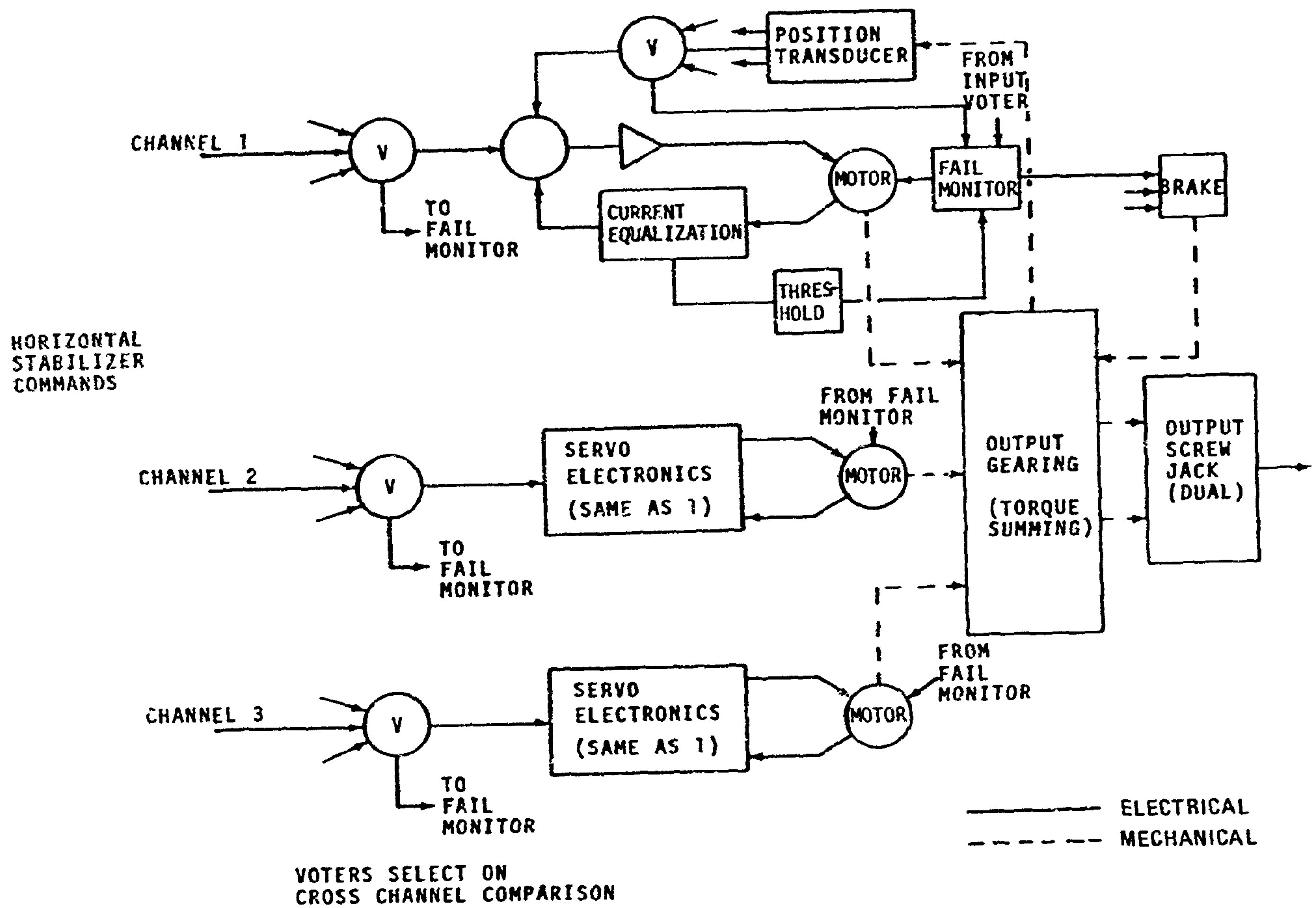

Figure A-23. Horizontal Stabilizer Actuator Control/Monitoring.

Cockpit Controls

Figure A-1 shows the arrangement of the pilot/copilot primary controls. Longitudinal/lateral control shall be provided by a two-axis force controller on the right seat arm. Directional control shall be provided by a conventional pedal arrangement. Collective pitch control shall be provided by a one-axis force controller.

Each force controller consists of a pilot's input device (control grip or pedals) working against a spring mechanism or flexure. Redundant optical transducers (one for each channel) are attached to the pilot's input device. Three transducers, each having a dual output to allow self-monitoring, are used to signal the triplex in-line monitored digital system. Each controller can output up to three serial words. The longitudinal/lateral force controller output contains the longitudinal control, the lateral control, and the longitudinal/lateral trim command. The directional force controller contains the directional control and provisions for the right brake, and the left brake outputs. The collective pitch control force controller contains the collective control, the collective trim command, and a spare word.

Pilot vernier (beep) trim input shall be provided by four-way switches located on the side controller for longitudinal/lateral trim and on the collective pitch lever for directional/collective pitch trim.

Table A-4 establishes the cockpit control forces and motions at the points of pilot control application.

REDUNDANCY MANAGEMENT

1. The PFCS shall be triplex. Each channel shall have two dual paths to allow self-monitoring. For each channel, the following failure detection circuits shall be available: actuator current monitor, EHV current with control stage position comparison, clock monitor, parity check of position transducers.

2. All failures causing loss of a primary flight control channel shall be detected immediately and displayed.

3. A failed channel shall be removed from the system as soon as necessary to maintain flight control operation.

4. The pilot and copilot controller output signals shall be mixed together and a disengage logic shall be available to shut off the pilot (or copilot) signal in case of hardover or force flight (Figures A-7 and A-8). A reset for this function should also be provided.

5. The AFCS shall be triplex. Any first failure or second failure detected by sensor voter or PFCS input voter shall be displayed by warning light.

6. Failure detection and warning/display logic shall be dualized where necessary to meet reliability requirements.

7. Reduction in vulnerability of one component shall not increase the vulnerability of another component or subsystem.

TABLE A-4. COCKPIT CONTROL - FORCE/DISPLACEMENT RANGE

FUNCTION	FULL-SCALE RANGE
LONGITUDINAL	$\pm$ 20 lb $\mp$ 0.200 in.
LATERAL	$\pm$ 10 lb $\mp$ 0.100 in.
DIRECTIONAL	$\pm$ 30 lb $\mp$ 0.50 in.
COLLECTIVE	$\pm$ 10 lb $\mp$ 20 in.

SYSTEM INTERFACES

Figures A-1, A-2, A-3, A-4, A-5, and A-6 show the overall interfaces between the flight control processor unit and other aircraft subsystems. These interfaces include:

Cockpit Controls

Mechanical interface with the cockpit controls. Fiber-optic force transducers or LVDTs shall be used to convert mechanical motions to optical signals.

AFCS

Digital interface with triplex AFCS. These triplex AFCS signals shall be voted before being mixed with the primary signals.

Main Rotors, Tail Rotor, and Horizontal Stabilizer

Mechanical interface with main rotors, tail rotor and horizontal stabilizer. There shall be two options:

1. LVDT shall be used to convert mechanical motions to analog signals. These analog signals shall be converted to digital signals in the processor unit.

2. Optical transducers shall be used to convert mechanical motions to optical signals. These optical signals shall be converted to digital signals in the processor units as described in Volume II.

Electrical Supply

Analog interface with regulated 28 VDC per MIL-STD-704B (Reference 12).

Hydraulic Supply

Hydraulic interface with hydraulic system providing 3,000 psi nominal pressure at all actuator required flows.

Interchannel Interface

Digital interface between channels shall be via dedicated optical links.

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12. Military Standard, MIL-STD-704A, Notice-2, ELECTRIC POWER, AIRCRAFT, CHARACTERISTICS AND UTILIZATION OF, Department of Defense, Washington, D.C., 5 May 1970.

CONTROL ACCURACY

Static Gain Accuracy

System gains shall be as specified in Figures A-7 and A-8. The average gain for all system channels shall be within 3 percent of the value specified. The static gain of individual system channels shall be within 2 percent of the average. For a given control input, the accuracy is defined as the percentage difference between the desired actuator position and the actual actuator position. These accuracies include schedule accuracies.

System Null

The total steady state null associated with the PFCS (sensor to actuator) shall not exceed 0.4 percent of actuator full stroke.

Resolution

Resolution is defined as the minimum change in control required to obtain actuator motion. The resolution shall not exceed 0.04 percent of actuator full stroke.

System Hysteresis

Hysteresis within the PFCS shall not exceed 0.08 percent of actuator full stroke.

Cross Coupling

28
B

Full motion of any axis or combination of axes shall not require more than 2 percent of full control force (in axes not in motion) to compensate.

FREQUENCY RESPONSE

The frequency response requirements apply to the actuators only. The rotor control actuator shall exhibit a second-order response with a natural frequency of 50 rad/sec and damping factor of 0.7. This response shall be achieved while driving a rotor load represented as a second-order response with a natural frequency of 45 rad/sec and damping factor of 0.7. This response shall be achieved with a tensile or compressive load of 1,260 pounds (dual system) while not exceeding a velocity of 3.00 in./sec. The frequency response of the horizontal stabilizer actuator shall be at least 3 Hz, 0.7 critical damping, for amplitudes up to 10 percent of total actuator travel.

FAILURE TRANSIENTS

The maximum acceptable control system transients due to any single failure in the PFCS shall be controlled by system tracking tolerances specified in the Static Gain Accuracy section

RELIABILITY

The reliability requirements in the categories of flight safety, mission, and maintenance malfunction shall be in accordance with Table A-5 for the flight control processor unit, Table A-6 for the cockpit controls, and Table A-7 for actuators. Flight safety and mission requirements are based on a 1.0-hour mission. Flight safety reliability is the probability of completing a 1-hour mission. Flight safety reliability is the probability of completing a 1-hour mission without loss of the vehicle. Mission reliability is the probability of completing a 1-hour mission without an abort. Maintenance malfunction reliability is the probability of not having a failure in a 1-hour mission. The numbers shown are on a per unit basis. They are allocations from the overall system reliability goals.

TABLE A-5. SUBALLOCATION FOR THE FLIGHT CONTROL PROCESSOR UNIT (ONE CHANNEL)

SYSTEM ELEMENT	FLIGHT SAFETY		MISSION		MAINTENANCE MALFUNC.	
	(FAILS/ MHRS)	RELIABILITY	(FAILS/ MHRS)	RELIABILITY	(FAILS/ MHRS)	RELIABILITY
Common Electronics (Failure affects 2 or more rotor positions simultaneously)	300	0.999700	300	0.999700	590	0.999410
Unique Electronics	35 (each of four)	0.999965	35 (each of four)	0.999965	30 (each of four)	0.999970
Total (1) FCP	440	0.999560	440	0.999560	710	0.999290
Pilot Control Panel	0	1.0	0	1.0	34	0.999966
Advisory Panel	0	1.0	0	1.0	37	0.999963
Sensor Mux/Test I/F	0	1.0	0	1.0	69	0.999931

TABLE A-6. SUBALLOCATION FOR THE COCKPIT CONTROLS

PILOT/COPILOT CONTROLS	QTY	FLIGHT SAFETY		MISSION		MAINTENANCE MALFUNC.	
		(FAILS/ MHRS)	RELIABILITY	(FAILS/ MHRS)	RELIABILITY	(FAILS/ MHRS)	RELIABILITY
Collective	2	0.099	0.9(7)010* (each of 2 positions)	0.25	0.9(6)75 (each of 2 positions)	600	0.999400 (each of 2 positions)
Longitudinal/ Lateral	2	0.099	0.9(7)010	0.25	0.9(6)75	600	0.999400
Directional	2	0.099	0.9(7)010	0.75	0.9(6)25	350	0.999650
Total Cockpit Controls		0.0(7)53	1.0	2.5	0.9(5)75	3100	0.996905

*0.9(7)010 = 0.9999999010

TABLE A-7. SUBALLOCATION FOR ACTUATORS (ONE POSITION)

FUNCTION	QTY	FLIGHT SAFETY		MISSION		MAINTENANCE MALFUNC.	
		(FAILS/ MHRS)	RELIABILITY	(FAILS/ MHRS)	RELIABILITY	(FAILS/ MHRS)	RELIABILITY
1/2 S/P Jam	2	0.0013	0.9(8)87*	0.0013	0.9(8)87	0	1.0
Function	2	7.539	0.9(5)246	7.539	0.9(5)246	200	0.99964
1/2 Driver Jam	2	0.00086	0.9(9)1	0.024	0.9(7)760	0	1.0
Function	2	34.4	0.9(4)656	34.0	0.9(4)656	400	0.99960
TOTAL		0.0071776	0.9(8)29	83.9286	0.999916	1200	0.99880

*0.9(8)87 = 0.9999999987

MAINTAINABILITY

The unscheduled maintenance requirements shall not exceed the following:

1. Main rotor actuators (all three actuators) — 1.39 MH/1000 FH
2. Tail rotor actuator — 0.34 MH/1000 FH
3. Flight Control Processor Unit — 5.76 MH/1000 FH

The above maintainability requirements are for organizational level maintenance (AVUM) where the primary maintenance activity is removal and replacement.

ENVIRONMENTAL REQUIREMENTS

Standard Conditions

The following conditions shall be used to establish normal performance characteristics under standard conditions for making laboratory bench tests.

1. Temperature – room ambient 25 ±5°C (77°F ±9°F)
2. Altitude – normal ground
3. Humidity – room ambient up to 90-percent relative humidity

Environmental Service

Components of the FCS shall meet the requirements of this specification under the conditions listed in the following paragraphs. Electronic components shall be tested under the conditions defined in MIL-E-5400T for Class 1A equipment. Actuators shall be tested to the conditions specified. The equipment supplier shall submit a detailed procedure to be approved by Boeing.

1. Altitude – Operation without degradation of performance throughout a pressure altitude range of –200 to +30,000 feet ASL per MIL-STD-810C (Reference 13).
2. Ambient temperature – Operation throughout an ambient temperature range of -65^{o}F to $+165^{o}$F.
3. Temperature shock – Sudden changes in temperature of the surrounding atmosphere per MIL-STD-810C.
4. Humidity – Operation in a warm, highly humid atmosphere such as encountered in tropical areas per MIL-STD-810C.
5. Salt Fog – Operation in an atmosphere containing salt-laden moisture per MIL-STD-810C.
6. Sand and dust – Operation in a dust- (fine sand) laden atmosphere per MIL-STD-810C.
7. Rain – Operation in a rain environment per MIL-STD-810C.
8. Immersion – (for hydraulic actuators only) Operation after immersion in hydraulic fluid at a temperature of $+275^{o}$F per MIL-A-5503D (Reference 14).
9. Vibration Operation during exposure to dynamic vibration stresses represented by those tests of MIL-STD-810C, Method 514.2, Procedure I, Part 1, Equipment Category C.
10. Mechanical Shock Operation after exposure to a mechanical shock environment similar to that expected in handling, transportation, and service use per MIL-STD-810C.

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13. Military Standard, MIL-STD-810C, ENVIRONMENTAL TEST METHODS, 10 March 1975.

14. Military Specification, MIL-A-5503D, ACTUATOR, AERONAUTICAL LINEAR UTILITY, HYDRAULIC, GENERAL SPECIFICATION FOR, Department of Defense, Washington, D.C., 15 June 1977.

ELECTROMAGNETIC INTERFERENCE

Equipments shall meet requirements defined in MIL-STD-461B (Reference 15), and, in addition, shall operate satisfactorily under threats defined in Reference 7.

FORM FACTOR

Table A-8 establishes the form factor requirements.

TABLE A-8. FLIGHT CONTROL EQUIPMENT FORM FACTORS

Factor	Flight Control Processor Units	Actuator
SIZE	4.88 in. wide x 7.62 in. high x 16.00 in. max depth (fiber optic - Figure A-24) 7.50 in. wide x 7.62 in. high x 12.52 in. max depth (LVDTS - Figure A-25)	Figure A-21
WEIGHT	14.9 lb max (fiber optic) 17.9 lb max (LVDTS) 2.5 lb - Control Panel 4.5 lb - Sensor/Mux Unit	28.3 lb max (main rotor) 24.0 lb max (tail rotor)
POWER	28 VDC per MIL-STD-704B	3000 psi Type II per MIL-H-5440C

15. Military Standard, MIL-STD-461B, ELECTROMAGNETIC EMISSION AND SUSCEPTIBILITY REQUIREMENTS FOR THE CONTROL OF ELECTROMAGNETIC INTERFERENCE, Department of Defense, Washington, D.C., 1 April 1980.

VENDOR DESIGN STUDY

TRADE STUDY CANDIDATES

The candidate systems for the trade study are defined in Figures A-24, A-25, and A-26, and shall comprise the following:

1. Baseline – In-line monitored triple model – fiber-optic transducers, (Figures A-2 and A-24).

2. Alternate – In-line monitored triplex model – LVDT, (Figures A-25 and A-26).

VENDOR EFFORT

The vendors shall be responsible for the following tasks as applicable in each of their respective areas of endeavor:

1. Mechanize concepts (baseline – optical transducers, alternate – LVDTs) defined in previous sections.

2. Provide preliminary design information including R&M factors and size, cost, and weight data on hardware according to the following schedule:

   a. Fleet size – 1,450 helicopters

   b. Production rate:

      (1) 2 helicopters per month for 6 months

      (2) 8 helicopters per month for 6 months

      (3) 15 helicopters per month for 4 months

      (4) 20 helicopters per month per remainder

   c. Aircraft life – 15 years

   d. Flying hour program – 480 hours per year

   These data will be used by Boeing to synthesize the overall system configuration.

3. Should any conceptual change in the candidate systems become definite in the vendor's opinion, such change shall be recommended to Boeing Vertol Engineering for disposition.

4. For technology purposes, the technology level selected for this work shall be available for inclusion in a prototype flight vehicle contract, which may be awarded in FY '80.

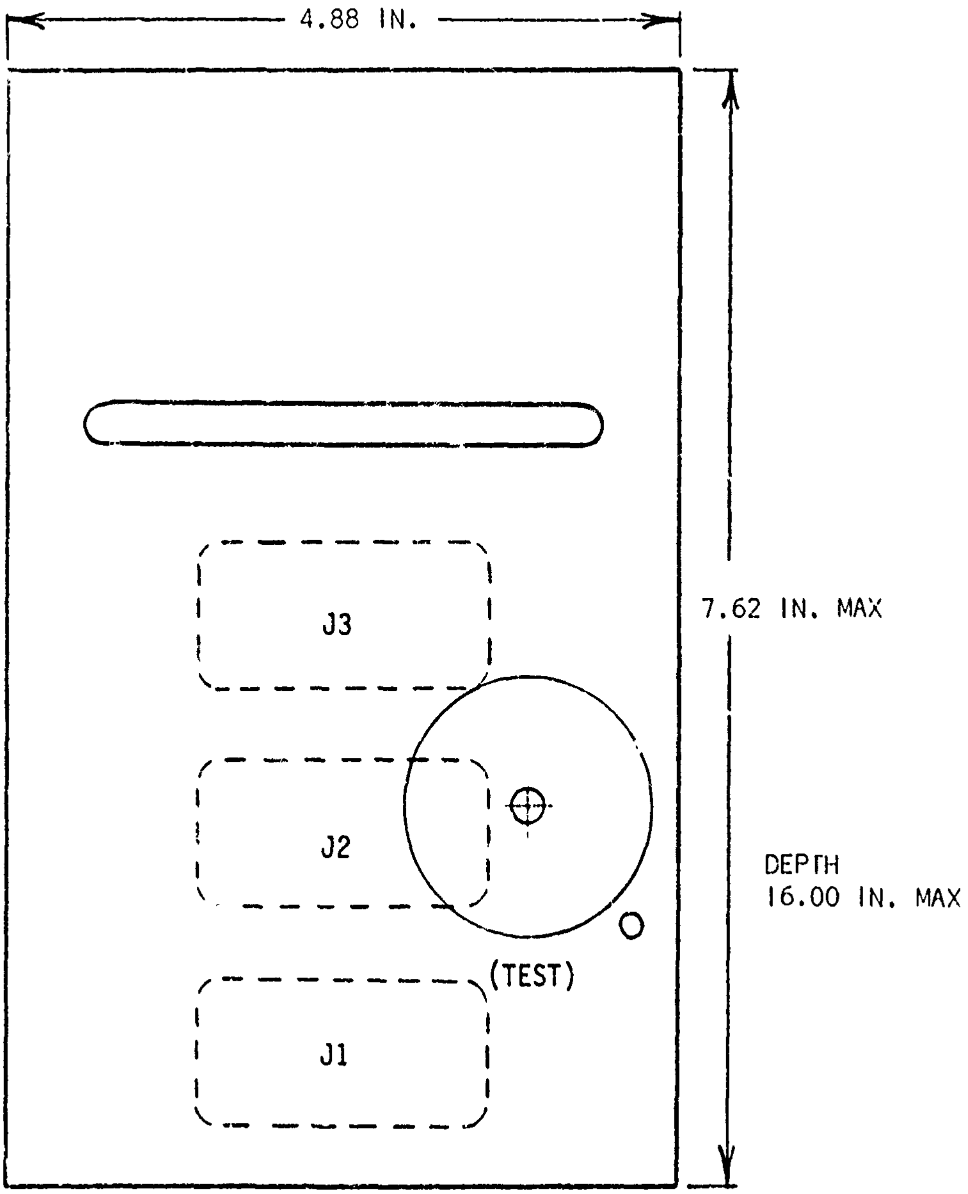

Figure A-24. Baseline Flight Control Processor - Outline.

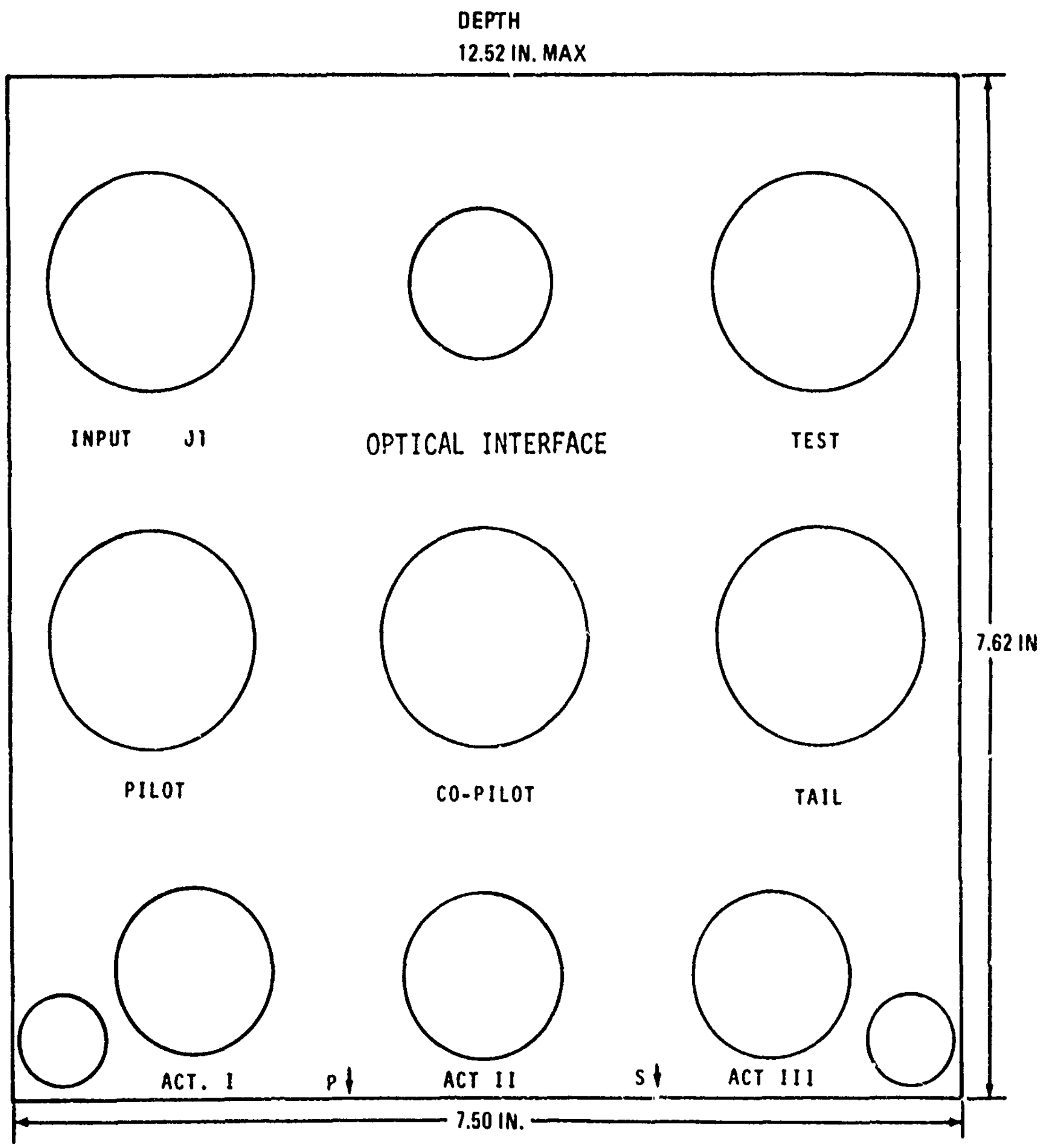

Figure A-25. Alternate Flight Control Processor - Outline.

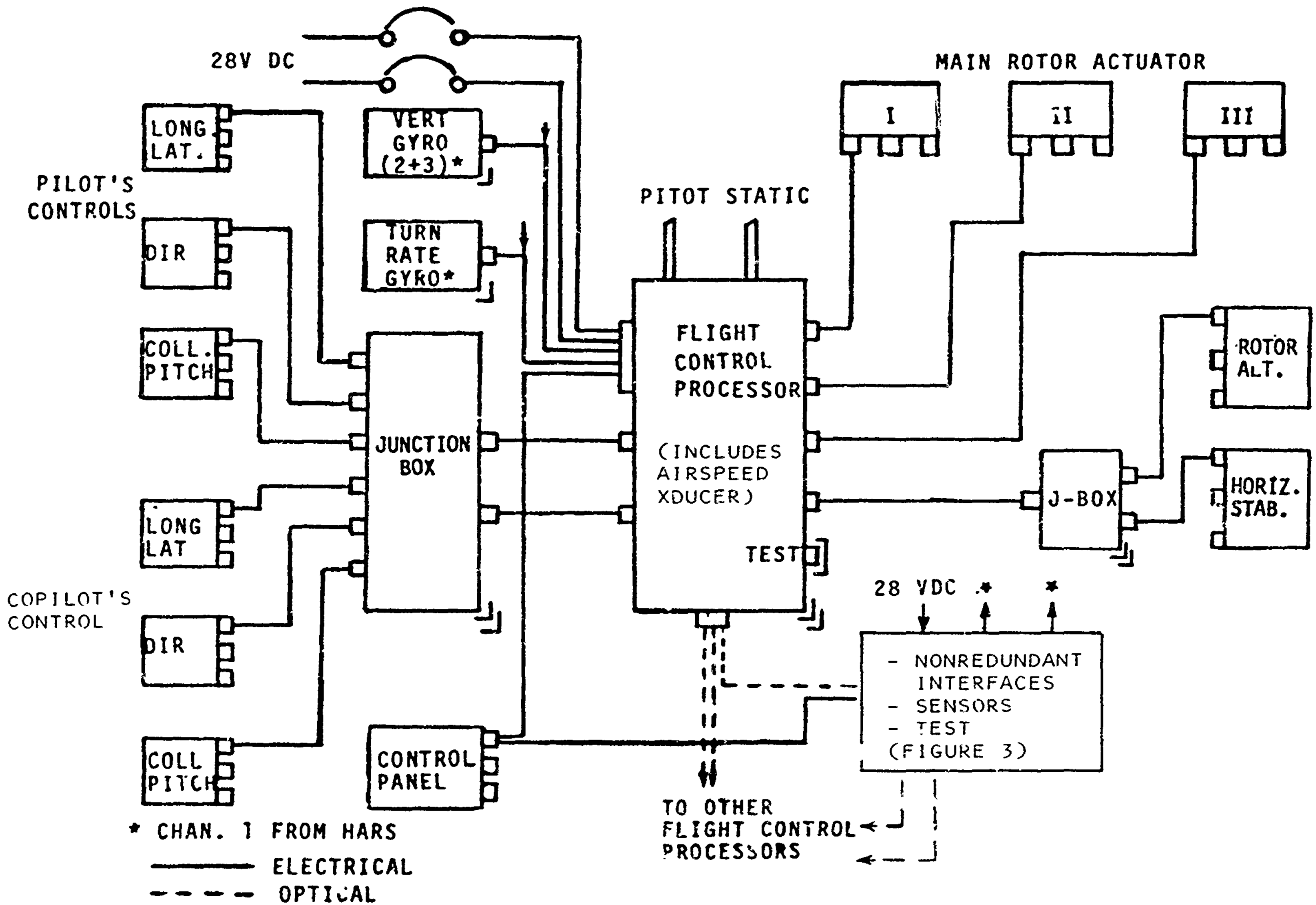

Figure A-26. Alternate ASH Flight Control System Equipment Diagram.

APPENDIX B

DEVELOPMENT OF CANDIDATE CONFIGURATIONS

This appendix describes the simulation activity undertaken to define the AFCS design and the feasibility and interface necessary for force-type controls, research activities on fiber optic transducers and cabling, and the refinement of candidate system configurations.

CONTROL CONCEPT DEVELOPMENT

A flight simulation activity was conducted to develop the primary and automatic flight control system concepts. Evaluation of flying qualities was performed in real time with a pilot-in-the-loop using the Boeing Vertol flight simulation facility. Major elements of the simulation facility include a Xerox Sigma 9 computer center and a flight simulator having a small motion capability and a closed circuit television/terrain model visual system. The terrain model is representative of the topography around Ft Rucker, Alabama, and is suitable for low altitude maneuvering tasks around trees and obstacles.

The BO-105 helicopter math model was used for the control law studies because it is operational and correlated with available flight test data. In addition, the BO-105 gross weight and size is comparable to an ASH type vehicle, and it provided typical single rotor stability and control coupling characteristics to be addressed during the flight control system design process.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) DESIGN

The pilot's ability to control the vehicle in the low speed/ hover flight regime was of primary concern during the simulation program. Typical low-speed pilot maneuvers used to evaluate flight control system performance included the short longitudinal dash, lateral junk, and vertical bob-up maneuvers. Angular rate, angular attitude, and linear velocity response systems were studied. Table B-1 summarizes the stability and control features developed for the ASH mission. The basic AFCS stability and control characteristics are defined for the low speed and forward flight regions, as well as selectable mode functions for hover hold and altitude hold. Mode selection logic in the AFCS design accomplishes engagement or disengagement of hover hold and altitude hold without control response transients. Logic is also provided for automatic transient-free switching between low-speed and high-speed flight control laws. The AFCS design specifications are documented in Appendix A.

The basic AFCS provides attitude hold stability for pitch, roll, and heading in all flight conditions. In forward flight, airspeed feedback and logic permits long-term airspeed hold and vernier airspeed trim capability in the longitudinal axis. AFCS logic is used in the lateral axis above 45 knots to provide bank angle hold about any trim bank angle condition and roll attitude beep capability with a trim button on the sidearm controller. The system design uses washed-out roll attitude feedback in hover and at low airspeeds to provide short-term stabilization while accommodating steady roll attitude trim requirements associated with wind changes. Heading hold with vernier trim capability is provided in the directional axis at all flight speeds. Low rate heading trim changes are commanded from a trim button located on the grip of the collective pitch controller. Altitude hold is included as a selectable mode designed to give vertical position hold based on either a barometric pressure or radar altimeter signal. Good altitude hold performance is achieved by combining the altitude reference signal with the vertical velocity signal available from the heading and attitude reference system (HARS). This complementary filter mechanization uses a low-pass filter on the altitude signal to eliminate unwanted high frequency sensor inputs (i.e., noise, spikes, gust upset), and a high-pass filter on vertical velocity to provide high frequency altitude stabilization while eliminating static long-term signal drift or offset.

TABLE B-1. ASH STABILITY AND CONTROL FEATURES

AXIS	BASIC AFCS				SELECTABLE MODES			
	HOVER/LOW SPEED		FORWARD FLIGHT		HOVER HOLD <45 KN		ALTITUDE HOLD (ALL AIRSPEEDS)	
	STABILITY	CONTROL	STABILITY	CONTROL	STABILITY	CONTROL	STABILITY	CONTROL
LONGITUDINAL	PITCH ATTITUDE	PITCH ATTITUDE	PITCH ATTITUDE & AIRSPEED HOLD	SHORT TERM- PITCH RATE PITCH ATT. LONG TERM- AIRSPEED	GROUND SPEED HOLD	SHORT TERM- PITCH RATE PITCH ATT. LONG TERM- LONGITUDINAL GROUNDSPEED	-	-
LATERAL	ROLL ATTITUDE	ROLL ATTITUDE	BANK ANGLE HOLD (ANY BANK ANGLE)	ROLL RATE WITH AUTOMATIC TURN COORDINATION	GROUND SPEED HOLD	SHORT TERM- ROLL RATE ROLL ATTITUDE LONG TERM- LATERAL GROUNDSPEED	-	-
DIRECTIONAL	HEADING HOLD	YAW RATE	HEADING HOLD (FOR ZERO TURN COMMAND)	YAW RATE	HEADING HOLD	YAW RATE	-	-
VERTICAL	VERTICAL VELOCITY	VERTICAL VELOCITY	VERTICAL VELOCITY	VERTICAL VELOCITY	ALTITUDE HOLD	VERTICAL VELOCITY	ALTITUDE HOLD	VERTICAL VELOCITY

The hover hold mode, also a selectable feature, gives a tight velocity hold about the helicopter longitudinal and lateral axis flight path. High velocity gains and the use of integral control loops provide pseudo-position hold capability to within approximately 1.0 foot under moderate gust conditions when sensor drift corrections are supplied by the pilot.

The hover hold mode engages radar altitude hold and retains the basic AFCS heading hold control loops.

FORCE CONTROLLER DEVELOPMENT

The low displacement, force-type control is one key to the development of a fly-by-wire control system. Application of a force control can allow a sizable cost and weight reduction, reduce controls vulnerability, and eliminate the reliability and maintainability problems of conventional cockpit mechanical controls. Feasibility of this helicopter control concept was successfully demonstrated using a Lear Siegler A-7/F-16 two-axis sidearm controller for longitudinal and lateral control. Figure B-1 is a picture of the sidearm controller installed in the simulator. The sidearm handgrip works against a spring mechanism or flexure, and has small redundant transducers attached to the controller in each axis to give an output signal proportional to force. The Lear Siegler controller evaluated has ±0.030-in. displacement of full scale and a maximum force capability of 25 pounds in each axis. The existing simulator collective and directional pedals were also converted to pure force controls rather than displacement controls.

AFCS ON control characteristics in forward flight were designed to provide angular rate response due to force control inputs for pitch, roll, and yaw. In the low-speed/hover flight regime, longitudinal and lateral force control inputs from the sidearm controller provide an initial pitch/roll angular rate response that blends to a near constant attitude response per pound of force. Nonlinear force response characteristics were necessary to achieve both optimum aircraft response for small precision maneuvering, and a larger roll and/or pitch attitude response desirable for the lateral jink and short longitudinal dash maneuvers. Assymetric force gradients in longitudinal and lateral control were also required to compensate for normal pilot arm and hand structure, and the natural tendency to push harder in one direction than the other. For instance, it is easier to apply a left force than a right force with a two-axis sidewarm force controller mounted to the right side of the pilot. The lateral force response characteristics obtained from the ASH simulation is presented on Figure B-2. This nonlinear and assymetric response shaping provided the pilot with a matched force-feel response characteristic for right and left control inputs and was acceptable for both low speed and forward flight.

The use of vertical velocity feedback to increase vertical damping provides a precise vertical velocity response characteristic due to force control input. A single-axis collective force controller is mounted to the left side of the pilot with a four-way trim button located on the grip to command small vertical velocity and low rate heading trim changes.

Control sensitivities and nonlinear force control gradient, shaping, and breakout characteristics were optimized for the AFCS OFF condition, and need to be tailored to the particular basic helicopter design. Optimum control responses with the AFCS on were then obtained by providing feedforward control command paths and logic within the AFCS to compensate the feedback control loops required for desirable levels of stability.

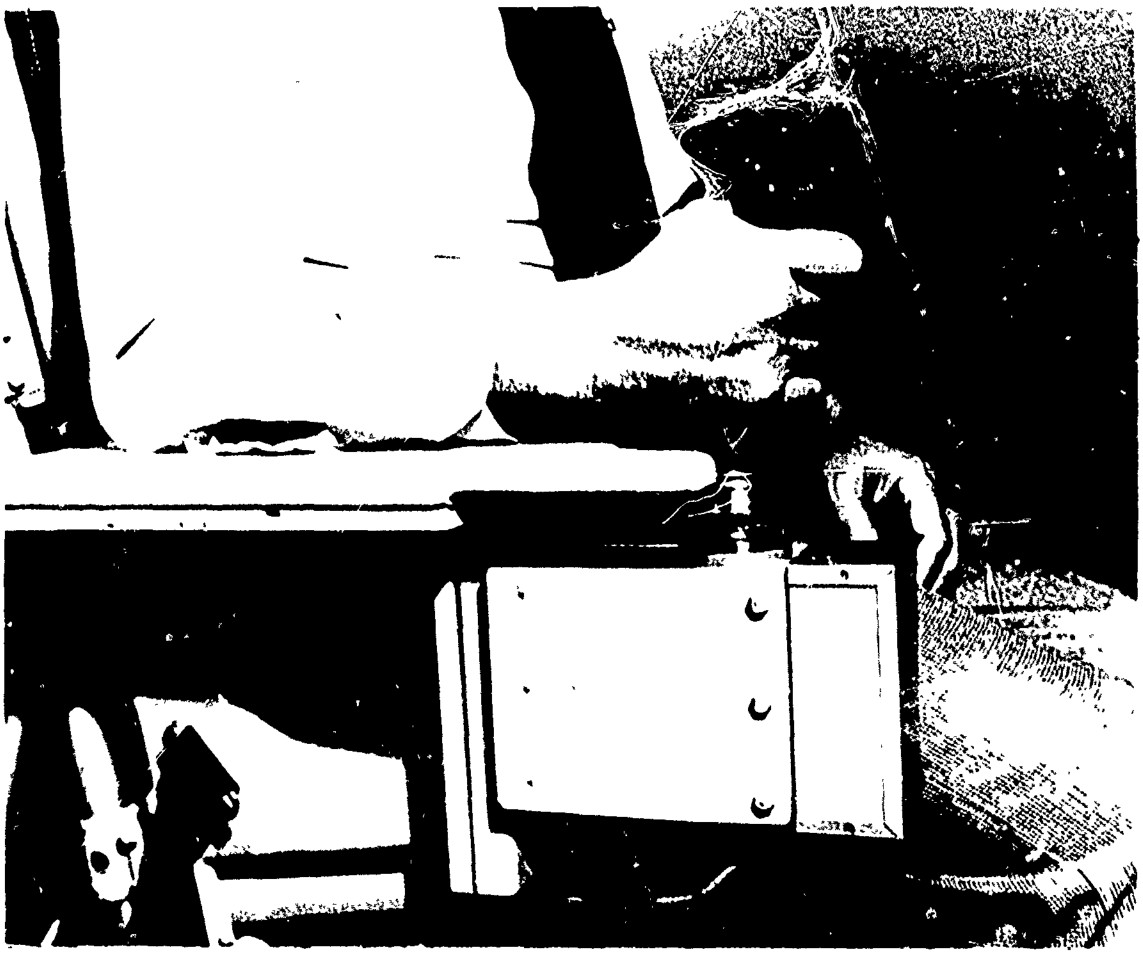

Figure B-1. Two-Axis Force Controller.

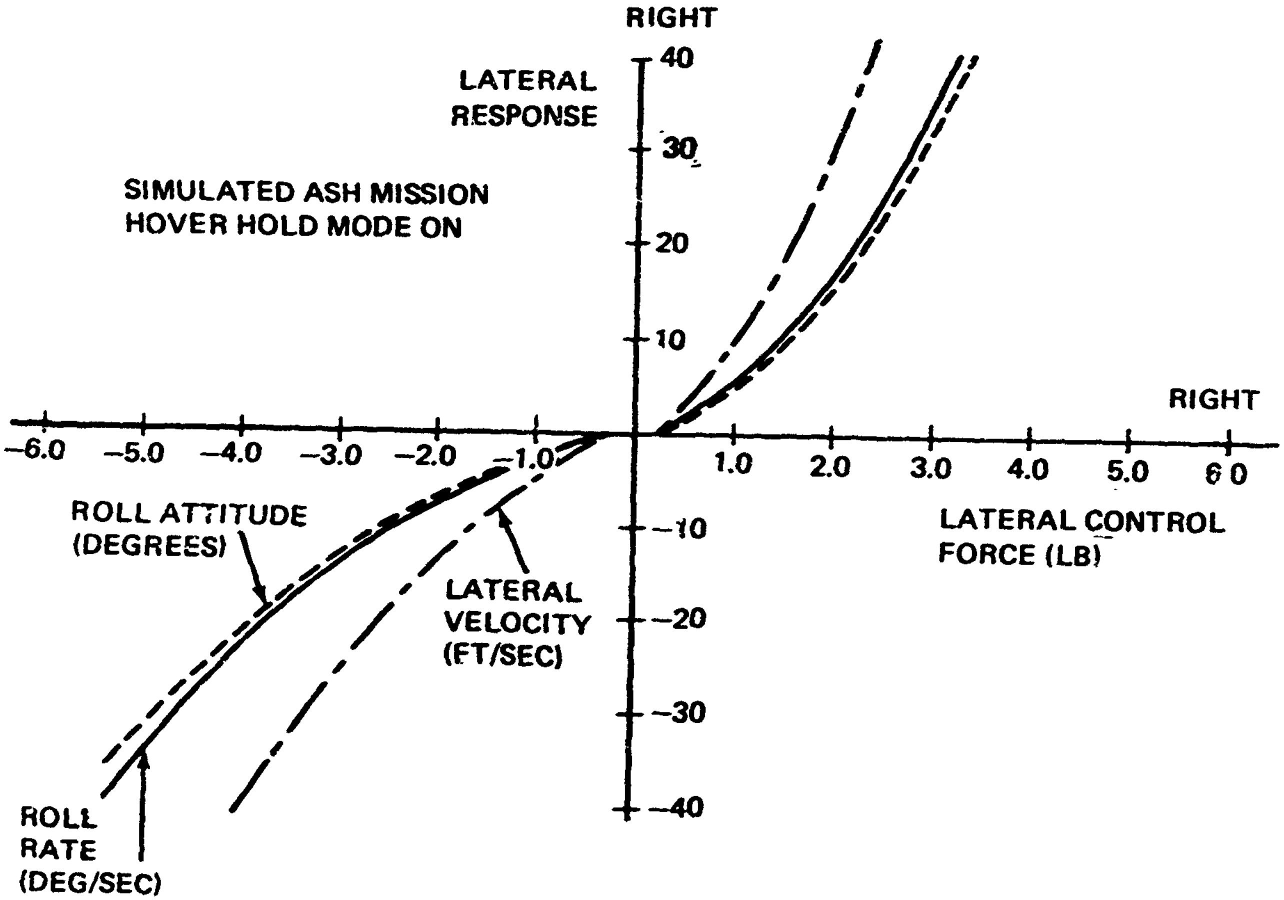

Figure B-2. Lateral Control Response.

The BO-105 helicopter could be flown with the AFCS OFF, but with a much higher pilot workload than with the AFCS ON, particularly for the hover control task. Flight simulator deficiencies for the hover flight regime contributed largely to this undesirable increase in pilot workload.

Many issues relating to acceptable force controller integration were investigated during the design and simulation activity. The following is a summary of results:

1. A suitable method was developed to permit trimming of controller forces to zero without control response transients. The control law design, however, minimizes any large buildup of controller trim forces for steady-state flight condition changes. The force trim method employs trim control logic implemented in the flight control processor (See Appendix A). When engaged in the hover hold mode, the controller can be referenced to a zero force condition for any desired longitudinal and/or lateral velocity trim condition.

2. The use of low-force trim switches is important to keep transients to a minimum or undetectable level. Switches requiring approximately 1.0 pound of force to activate were installed on the longitudinal/lateral sidearm and collective level controllers.

3. Capability to disengage and reset either the pilot or copilot force controller is incorporated in the primary control path to protect against hardover or inadvertent inputs. Each pilot requires a disengage button on his controller to disable the other pilot force control channel and ramp the unwanted force command input signals to zero at a slow rate. The disabled force controller can be reset and engaged again only if specific signal level and logic criteria for reengagement are satisfied.

4. The position and orientation of a two-axis sidearm controller and a single-axis collective thrust controller is important to prevent unintended coupling forces due to arm and hand structure, and to compensate for natural tendencies to push harder in one direction than the other. Requirements for a force controller grip type, mounting, and arm rest design were studied and a suitable configuration was defined based on piloted simulation results.

OPTICAL TRANSDUCER DEVELOPMENT

Details on Boeing's proprietary displacement transducer concept are given in Volume II of this report. The concept originally proposed by Boeing required one emitter for each transducer bit and multiple input/output lines for each transducer. The new scheme allows use of one emitter per line replaceable unit (LRU) (i.e., controller or actuator) and allows all transducers within the LRU to be accessed using one input fiber and two output fibers.

The newly developed transducer concept has been shown to be feasible although not completely demonstrated in Boeing tests. The demonstration model was designed around available standard optical fibers and components. As a result, losses were greater than would be expected in a production design that would incorporate customized components. Given adequate development funding, the transducer development risks can be minimized.

The resulting transducer designs are comparable with present electrical devices in size and cost.

FIBER-OPTIC CABLING

During the study Boeing discussed system cabling requirements with several suppliers of fiber-optic cable and connectors.

For the system cabling, a three-conductor design similar to that developed for Boeing Commercial Airplane Company (BCAC) per Boeing Specification 280-38001 (Reference 16) was chosen. This cable is manufactured by Galileo Electro-Optics Corp. Because of the short cable lengths used in an aircraft, it appears that signal loss per unit length is not a significant factor. Suitability of the fiber cable assembly to vibration, temperature, radiation, handling, and compatibility with connectors appear to be the most significant factors for the ASH application.

The status of connector development was reviewed with Amphenol, ITT Cannon, Deutsch, and Amphenol (906 series) and ITT Cannon (PV series) are developing the multiple contact metal connector needed for the baseline system design. Amp is developing only plastic units at this time. Deutsch has a low-loss single-fiber connector, which is not suitable for the baseline design. In the near future, they may develop a smaller version of the unit suitable for multiple conductor connectors. Data from Hughes (C-21 series per MIL-C-85028 (Reference 17)), supplier of rectangular multiple conductor connectors to BCAC, was also reviewed. Hughes connectors would also be considered in a follow-on hardware design.

Most suppliers quote a 1-2 dB loss per connection, which is compatible with the Boeing transducer concept. Currently the optical contact size is similar to that designed for a number 12 wire. Future designs may fit into a number 16 or even 18 size contact. The allowable contact size depends in part on the size of the selected optical cable assembly (including its buffer, strength, and abrasion members.)

For the baseline design, ITT Cannon connectors were selected. These units consist of standard electrical designs (number 12 contact size) into which the Cannon fiber-optic contacts are inserted. For the electronic LRU, the DPKA and DPKB designs per MIL-C-83733 are used in a back-mounted rack and panel design. For controllers and actuators, a circular threaded design per MIL-C-83723 (Reference 18) is used.

The cable and connectors selected here are considered suitable for initial design and costing of the system. A more comprehensive evaluation of the contending products would be required in any hardware development phase. Based on evaluations to date it appears that suitable cable assemblies can be developed at low to moderate technical risk.

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16. Mulky, O. R., SINGLE FIBER, FIBER OPTIC CABLE, Specification 280-38001, Boeing Aerospace Company, Seattle, Washington, 22 March 1978.

17. Military Specification, MIL-C-85028, CONNECTOR, ELECTRICAL, RECTANGULAR, INDIVIDUAL CONTACT SEALING, POLARIZED CENTER JACKSCREW, GENERAL SPECIFICATION FOR, Department of Defense, Washington, D.C., 22 October 1977.

18. Military Specification, MIL-C-83723D, CONNECTOR, ELECTRICAL (CIRCULAR, ENVIRONMENT RESISTING), RECEPTACLES AND PLUGS, GENERAL SPECIFICATION FOR, Department of Defense, Washington, D.C., 27 December 1977.

CANDIDATE CONFIGURATION DEVELOPMENT

BASELINE SYSTEM

The Boeing proposal described a baseline electro-optic approach and three alternatives. The baseline system (Figure B-3) is triple redundant, in-line (self) monitored. Each channel receives output from the pilots' controllers and a vertical gyro. Each channel includes the following electrical/electronic units: a sensor unit, flight control processor, actuator processors, and a power converter. Each channel also receives inputs from a set of nonredundant sensors to provide altitude hold, heading hold, and ground speed/position hold. These are shared with other systems.

The primary and automatic portions of the flight control system are processed in the same computer. The primary system is designed to be fail-operative for two similar failures in the control sensors, processing and power supplies; and single-fail operative for certain remote actuator failures. The AFCS is designed to give fail-operative, fail-off performance for basic stabilization and attitude retention functions and fail-off (with minor transient) for nonredundant functions.

The proposed baseline configuration employed electronics at or near the sensors to minimize the number of optic lines between the remote sensors and the flight control processor. The fiber-optic sensor used in the proposal required one input line per bit and two output lines or 26 lines for a 12-bit actuator position sensor. In addition, the scheme required one emitter per bit. These limitations led BAC to the development of the sensor discussed in Volume II of this report. The new scheme required only one emitter per LRU (i.e., actuator or controller) and three fibers per LRU – one for input and two providing a redundant output. This development led to the current baseline condiguration (Figure B-4), which eliminates the remote electronics.

The electrical lines going to the rotor actuators (two twisted, shielded pairs) provide current drive to the actuators electrohydraulic valves. These are analog signals and can be adequately protected from lightning and electromagnetic pulse (EMP) by use of shielded wiring. This approach precludes use of electronics near or on the actuator, which enhances the performance and reliability of the system. If a pure optical control is shown to be required by the EMP threat, it can be achieved by (1) use of an optical/hydraulic valve, which may be developed in the future, or (2) locating electronics (to process the EHV command and return the current level to the FCP) on the actuator. Power for the electronics located on the actuator would be from the system 28 VDC supply. Protection of this input would be accomplished by shielding and filtering on the actuator. An alternative approach would be to generate electrical power on the actuator. This appears to be unnecessary complication and degradation of system reliability.

Refinement of the original scheme also resulted in:

1. The elimination of the data bus (for interchannel communication) in favor of dedicated two-way fiber-optic links to connect the channels to each other and to the sensor multiplex/test interface unit.

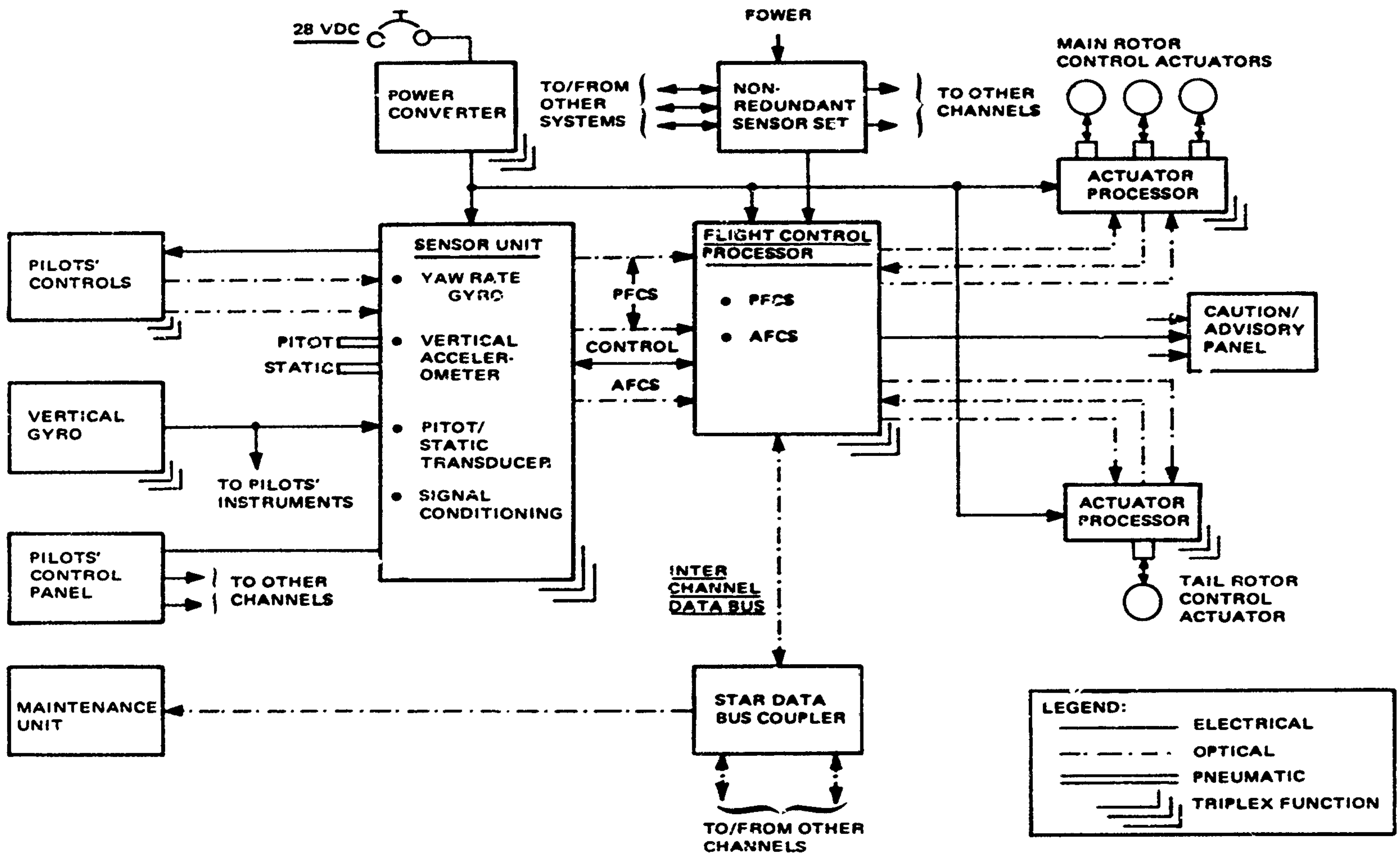

Figure B-3. Proposed Baseline Flight Control System.

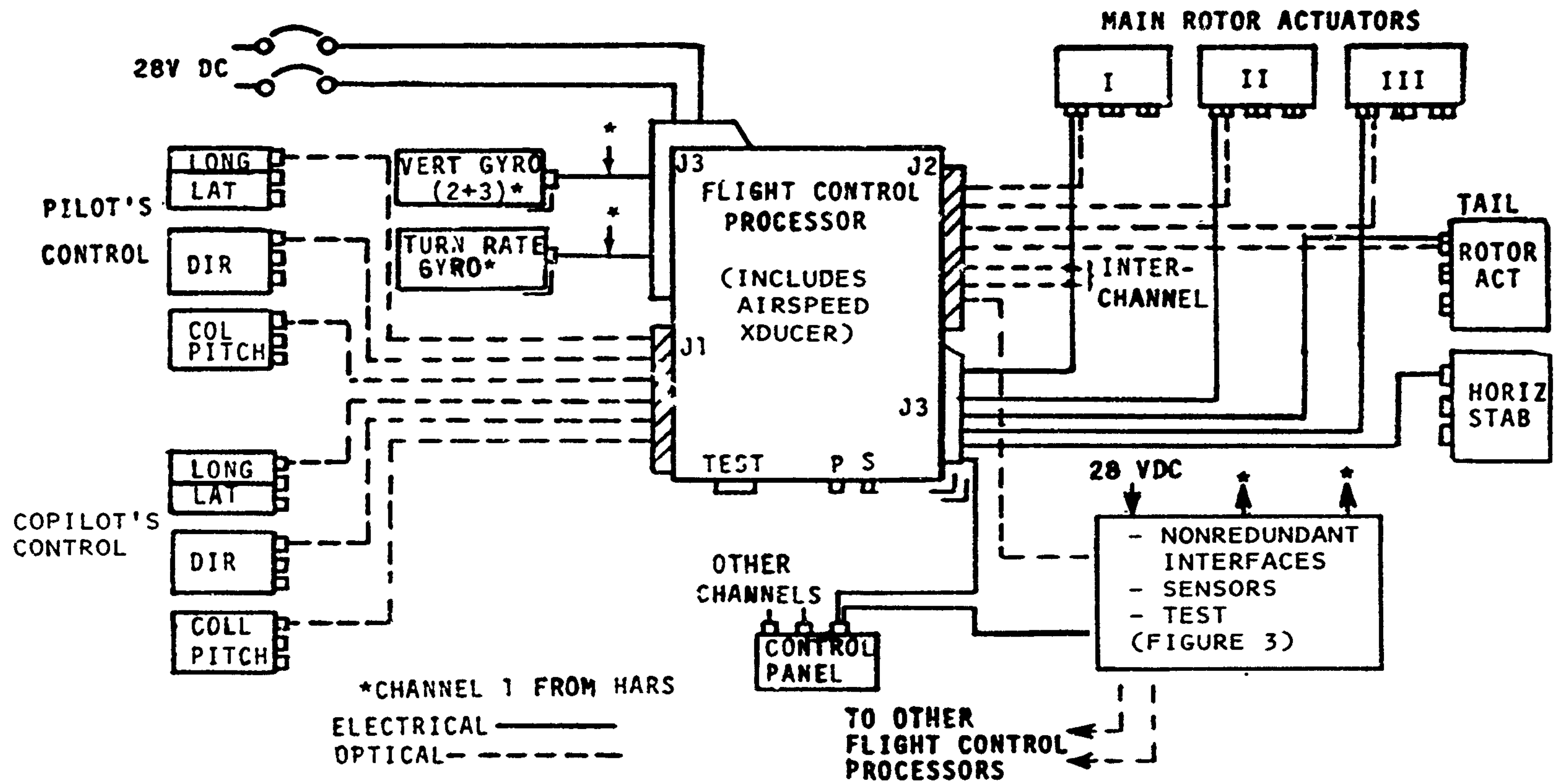

Figure B-4. Baseline ASH Flight Control System - Equipment Diagram.

2. A revision to the actuator monitoring technique, which eliminated two differential pressure transducers while adding one control stage feedback.

The final configuration, shown in Figure B-4 and described in the fly-by-optics section, includes six pilot control transducers, three flight control processors, four rotor control actuators with a noncritical horizontal stabilizer actuator, the sensor/multiplex/test interface unit, and control panel. The only AFCS sensors charged to the system are: one airspeed transducer per channel located in the flight control processor, and one static transducer for barometric altitude hold in the sensor multiplex/test interface unit.

ALTERNATIVE SYSTEM

In addition to the baseline configuration described above, Boeing proposed to evaluate the following alternatives:

1. An analog approach using conventional cockpit controls, analog processing, LVDTs, and no multiplexing. This is similar to that proposed for UTTAS. This scheme was dropped because it was deemed to be noncompetitive in the production ASH time frame. If the validation of a digital primary system (as recommended in the final baseline and alternative schemes) cannot be achieved with adequate confidence, use of an analog PFCS integrated in the same LRU with a digital AFCS would be the next best alternative. In this case, large scale integration (LSI) techniques would be used to reduce the cost, weight, and size of the analog portion.

2. A microprocessor-based digital approach with primary and automatic processing in the same computer, using force-type controls, dedicated wiring for channel communication, and multiplexing for interchannel communication using fiber-optic links. This scheme was retained as the alternative configuration. It is described in the Fly-by-Wire section of the main text.

3. A multiplex system similar to the baseline except that conventional LVDTs will be used in place of linear fiber-optic transducers. This configuration was also dropped from consideration, based on the judgement that the cost and weight of the remote electronics would more than offset the control weight of the dedicated wiring used in the selected alternative scheme.

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